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International Ultraviolet Explorer (IUE) Battery History and Performance

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PREFACE

The batteries for the "International Ultraviolet Explorer" (IUE) spacecraft was an in-house project in that they were designed, fabricated, and tested (Qualification and Acceptance) at the Goddard Space Flight Center. Because of the proper management of battery and of the other subsystems, the IUE spacecraft produced the largest amount of quality scientific publications over its 18-plus years of operation.

This report provides the information on the cell/battery design, the battery thermal characteristics, power system description, the third electrode charge control scheme, the battery performance during the 38 solar eclipse seasons and the end-of-life test data.

The "Power System Description" section presents the design goals, thermal characteristics, and the description of the subsystem - batteries, solar arrays and power system electronics. The "Cell Development Program" and "Cell Manufacturing" sections discuss the unique design changes, mandatory inspection points, and the acceptance and the characterization test data. The "Battery Design" section describes the conceptual design, packaging, and the electrical interface of the battery. The "Battery Fabrication" section presents an overview of mechanical and electrical assemblies, and the acceptance test and environmental test data of the battery. The "Spacecraft Performance Environmental Test" section provides the special test data which were conducted with batteries integrated into the spacecraft to ascertain the thermal properties of the batteries during the thermal vacuum and thermal balance tests. The "Space Flight Performance" section describes the prelaunch preparations and the battery operational guidelines, and provides pre-launch, launch, and in-orbit (38 eclipse seasons) performance data. The "End-of-Life Operations" section presents the data on the special tests that were conducted prior to the termination of spacecraft operation to check the battery switching operation, battery residual capacity, third electrode performance and battery impedance. The "Conclusions" section identifies the proactive actions that were implemented during the cell/battery design, and as a successful battery management over the eighteen plus years of the mission life. The final section, "Lesson Learned," identifies the areas for the improvement in the future battery design and for the in-orbit battery operation.

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LIST OF ACRONYMS AND ABBREVIATIONS

Ah	Ampere hours
C	Rated capacity (nameplate capacity)
°C	Degree Celsius
DOD	Depth of discharge
EOC	End of charge
EOD	End of discharge
gms	grams
GE	General Electric
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
I&T	Integration and Test
IUE	International Ultraviolet Explorer
KOH	Potassium Hydroxide
KSC	Kennedy Space Center
MCD	Manufacturing Control Document
NASA	National Aeronautics and Space Administration
NiCd	Nickel-Cadmium
NSWC	Naval Surface Warfare Center, (Crane, Indiana)
NWSC	Naval Weapons Support Center, (Crane, Indiana)
OC	Orbit condition -Sequence - Cycle
Psi	Pounds per square inch
QA	Quality Assurance
S/N	Serial Number
UMB	Umbilical

1.0 INTRODUCTION

The International Ultraviolet Explorer (IUE) was a joint project of three agencies: the National Aeronautics and Space Administration (NASA), the European Space Agency (ESA), and the United Kingdom (UK) Science and Engineering Research Council (SERC). IUE was launched on January 26, 1978, into an elliptical geosynchroncous orbit with a time period of 23 hours 58 minutes, eccentricity 0.2 inclined at 28° to the equatorial plane. The spacecraft contained a 45 cm Ritchy-Chretien telescope and two echelle spectrographs which were used to study the solar system, stellar, interstellar, galactic, and extragalactic astronomy. The spacecraft was powered by two solar array panels and two six ampere-hour (Ah) batteries.

This report provides a record of events of the IUE batteries from the initial design concept to space flight of 38 eclipse seasons of 24–30 days. This achievement surpassed the initial design life of 3 years by 15 plus years for a total of 18 and 1/2 years of operations. An enormous amount of data were acquired during that period, of which the authors summarize major events and analysis. This information highlights activities including cell design and manufacturing goals, cell acceptance, life test, and accelerated mission simulation test. Information derived from these studies were used for cell selection for the flight batteries, spare batteries, flight-spare cells and life-test cells. The document examines battery structural design, battery assembly, and battery preload. Brief comments are provided to highlight environmental test i.e., magnetic, thermal vacuum, thermal balance and vibration followed by the Goddard Space Flight Center (GSFC) in-house final acceptance test. Batteries Serial Number's (S/N) 03 and 04 were selected as flight batteries, and S/N's 05 and 06 as flight spares. Batteries S/N's 01 and 02 were identified for the IUE program as integration and test batteries, which are not covered in this document. After integration into the spacecraft, the flight batteries were closely monitored to determine their flight worthiness. During the spacecraft prelaunch checkout, a decision was made to replace the flight batteries SN's 03 and 04 with the flight spare batteries SN's 05 and 06. The decision was based on several factors: questionable cell third electrode low signal amplitude, temperature deltas observed during thermal vacuum and thermal balance, and the fact that the flight batteries had been exposed to 6 months of spacecraft acceptance test (considered excessive by the Power Branch). Summaries of battery data acquired during prelaunch, launch, transfer, park and in-orbit (Eclipse Seasons 1 through 38) are included with comments and conclusions.

2.0 POWER SYSTEM DESCRIPTION

2.1 BATTERIES

The phase B of the IUE project spacecraft design concept specified two 12 Ah Nickel-Cadmium (NiCd) batteries with each battery containing 17 cells to support daily eclipse periods during biannual eclipse seasons of 24 to 30 days each. During the early phases of hardware development, it became evident that the total spacecraft weight would exceed the launch vehicle capability. Consequently a series of design tradeoffs were studied. The mission orbit was reconfigured to a modified geosynchronous orbit to permit more sun time and less battery usage, After several design iterations, a decision was made to reduce the battery weight by using two 6 Ah NiCd batteries containing 17 cells each. Pertinent battery design characteristics down sizing the batteries from 12 Ah to 6 Ah are summarized as follows:

- Available battery power to the spacecraft reduced from 164 W/battery to 82 W/battery,
- Maximum battery discharge current reduced from 8 A/battery to 4 A/battery,
- Battery maximum depth-of-discharge (DOD) increased from 50 percent to 80 percent,
- Battery weight decreased from 23.10 lbs/battery to 12.76 lbs/battery,
- Battery size reduced from 470 cubic inches/battery to 280 cubic inches/battery, and
- Maximum battery charge current reduced from 1.2 A to 0.6 A.

After several design iterations, the final design of the 6 Ah battery and battery interface with the spacecraft was formalized.

2.2 SPACECRAFT BATTERY THERMAL CHARACTERISTICS

The two 6 Ah batteries were mounted adjacent to each other with one battery being inboard and the other outboard respective to the spacecraft center line. The selection of this concept was to accommodate available space and be compatible with the spacecraft harness configuration. (See figure 1). Thermal analysis indicated that the batteries would have good thermal dissipation. Heat dissipates through two heat pipes on the main spacecraft equipment platform directly under the batteries and through the louvers adjacent to the outside battery looking into deep space. Thermal data acquired from the spacecraft thermal models predicted that the battery's orbital temperatures would average 10 °C.

- Manually programming the charge current to a high or low rate 0.3-0.2 or 0.2-0.1 A respectively.
- Programming the main chargers on with the third electrode signals switched on.

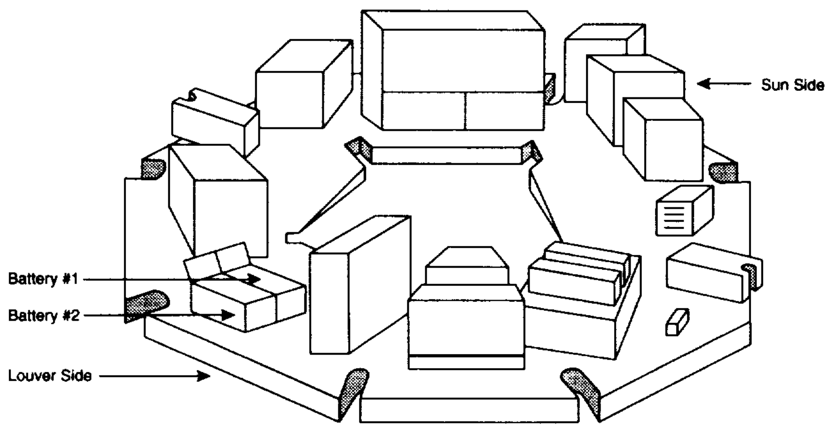


Figure 1. Main equipment platform.

2.3 POWER SYSTEM ELECTRONICS

A brief description of the Power System Electronics (PSE) is given to provide some of the critical control parameters used to maintain and operate the batteries during ground testing and space flight. The battery charge operations were controlled by individual control units of the PSE system. The system is a Direct-Energy-Transfer (DET) system operating at a bus voltage of 28± 0.056 V. Bus regulation during sunlight was accomplished through shunt dissipators. During eclipse periods or anytime the space-craft power demands exceed the solar array output, power was supplied by the batteries through a boost regulator operating at 90 percent efficiency. (See figure 2). Battery recharge current was controlled at a C/10 (0.6 A) rate. Control of the charge operations was accomplished by manually programming one of three charge-control systems:

- The main charger with the 3rd electrode signals switched off - the charge current will enter taper charge when a battery's terminal voltage reaches 25 V.

The third electrode control provides 0.6 A of battery charge current whenever the third electrode signal is less than 0.15 V and provides a current taper from 0.6 A to 0 A as the third electrode signal increases from 0.15 V to 0.24 V. However, in normal operations the battery current is never reduced to 0 A due to the third electrode characteristics, which calls for some charge current to maintain the third electrode signal level, a general characteristic of most closed loop systems. The third electrode threshold level of 0.15 V with a bandwidth of 0.09 V was determined after analyzing third electrode characterization test.

2.4 SOLAR ARRAYS

The primary source of power for the spacecraft was generated by two solar array panels. For the launch and transfer orbit periods, it was estimated the arrays would supply approximately 107 W of power to the bus when the spacecraft is in a stowed configuration. After deployment the maximum power expected at the beginning of life and winter solstice was 432 W. This is in compliance with the mission orbital requirements over the expected spacecraft beta angles of 0° through 135°.

greater than 4cc/Ah; and the use of nickel braze cell seals. Cells containing this type seal were manufactured in 1969 and tested in 1971 at temperatures ranging from -20 °C to 40 °C. No seal failures were noted during the tests.

Final design of the GE IUE 6 Ah cells incorporated the following features:

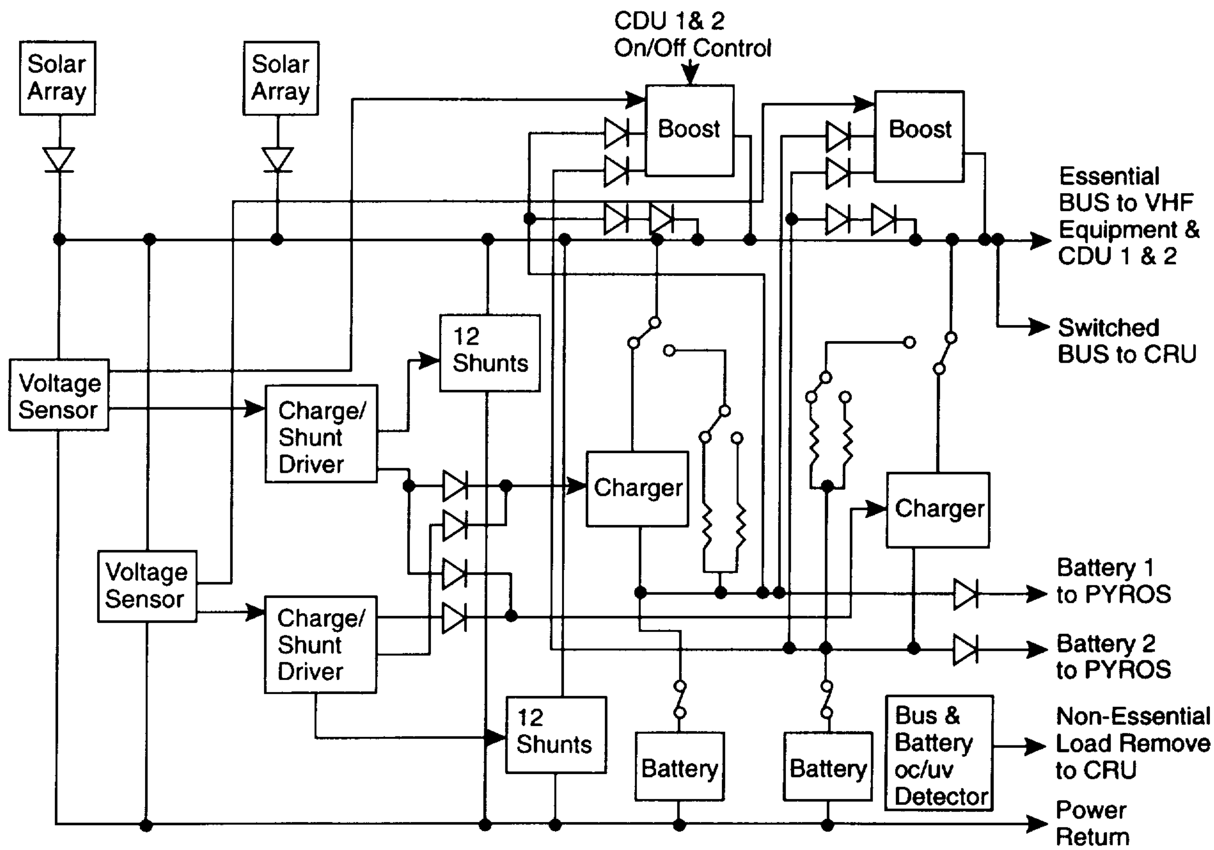


Figure 2. Spacecraft power subsystem.

3.0 CELL DEVELOPMENT PROGRAM

Data acquired from cell life test conducted at the Navy Ammunition Depot (NAD) in Crane, Indiana and from characterization test conducted at GSFC in Greenbelt, Maryland supplemented the IUE cell development and design program at the cell manufacturer.

During the initial cell procurement development in 1973, several technological advancements were studied and incorporated in the procurement of the 6 Ah cells from General Electric (GE), Gainesville, Florida. They were reduction in active cell plate material loading; use of teflonated negative, potassium hydroxide (KOH) quantity

- Ceramic-to-Metal Seal on positive and negative terminals
- Pellon 2505 separator,
- Teflonation of negative plate, level 1,
- Carbonate reduction process,
- P.Q. plate with light loading (Goal: 10% reduction in loading),
- Higher quantity of KOH (Goal: 4cc/rated Ah),
- Positive plates (0.026" to 0.028" thick),
- Negative plates (0.031" to 0.032" thick), and
- Drawn stainless containers.

4.0 CELL MANUFACTURING

The spacecraft environmental conditions and battery power requirements were used to define and establish design goals for the nickel-cadmium battery cells. Important parameters were the capability of the cells to reach a full charged condition within a prescribed time period and the production of the required output power at a defined voltage potential while maintaining a minimum internal pressure and temperature. It was equally important that these conditions be satisfied over the specified 3-year design life and 5-year design goal.

4.1 CELL MANUFACTURING DATA (6 Ah NiCd)

The data listed below are the results obtained during cell manufacture. It should be noted that the increase in the amount of electrolyte (31% KOH) and light plate loading resulted in a level of 4.17 cc/rated.

• Loading - positive average (g dm^{-2})		12.72
	negative average (g dm^{-2})	16.2
• Theroretical capacity - positive (Ah)		10.13
	negative (Ah)	18.19
• Flooded cell tests - positive average (Ah)		7.81
	negative average (Ah)	14.48
• Negative/Positive ratio		1.85
• Precharge set (by 0_2 venting) (Ah)		2.84
• Electrolyte (31% KOH) (cc/rated Ah)		4.17

4.2 CELL CHARACTERIZATION

In conjunction with the NiCd cell procurement program, the cell characterization test was conducted on a test pack containing five 12 Ah cells by GSFC from 1974 through 1975. The following test parameters were evaluated during the test period.

- charge control parameters and
- cell degradation.

After the 527 cycles (C/20 charge for 23 hours and C/2 discharge for 1 hour), at 50 percent Depth-Of-Discharge (DOD), the test pack delivered over 11 Ah of capacity when discharged to 1.0 V first cell.

The life test of the 12 Ah cells was conducted at NAD Crane, Indiana at three temperatures of 0, 10, and 20 degrees celsius (°C) using the IUE charge control profile (C/20 charge 23 hours, C/2 discharge 1 hour), 50 percent DOD. After 1.5 years of testing, capacities delivered were 14.2, 13.1, and 11.4 Ah respectively.

4.3 GSFC 6 Ah CELL PACK TEST

A 5-cell test pack (Pack No. 3) was fabricated with the GE 6 Ah cells and then was characterized at GSFC. The program was conducted in accordance with procedures described in the "GSFC Test Plan for IUE Prototype Cells"— TP 711.2-74-02. Additional tests and modifications to the test plan were implemented to provide accelerated test data to support the IUE spacecraft program. The test outline for the cells is as follows:

- Conditioning,
- Capacity (baseline),
- Open Circuit Voltage Recovery,
- Overcharge,
- Third Electrode Characterization,
- Cycling,
- Post Cycling Capacity,
- Accelerated Eclipse Season Simulation,
- Post Eclipse Season Simulations Capacity, and
- Open Circuit Voltage Recovery.

The test pack (No. 3) was initially conditioned by charging it for 48 hours at a constant current rate of 0.3 A (C/20) and at 20± 2 °C. The test instrumentation (battery voltage monitor) was set to 7.5 V (1.5 V/cell) to provide voltage cutoff protection.

Throughout the test program capacity testing was conducted at 0, 10, and 20°C to provide data to compare with the initial baseline capacity and eclipse season simulations data. The capacity curves (0, 10, and 20 °C) at the beginning of the test are shown in figure 3. Figure 4 shows a simplified third electrode-control circuit used to control a battery's recharge prior to a capacity test.

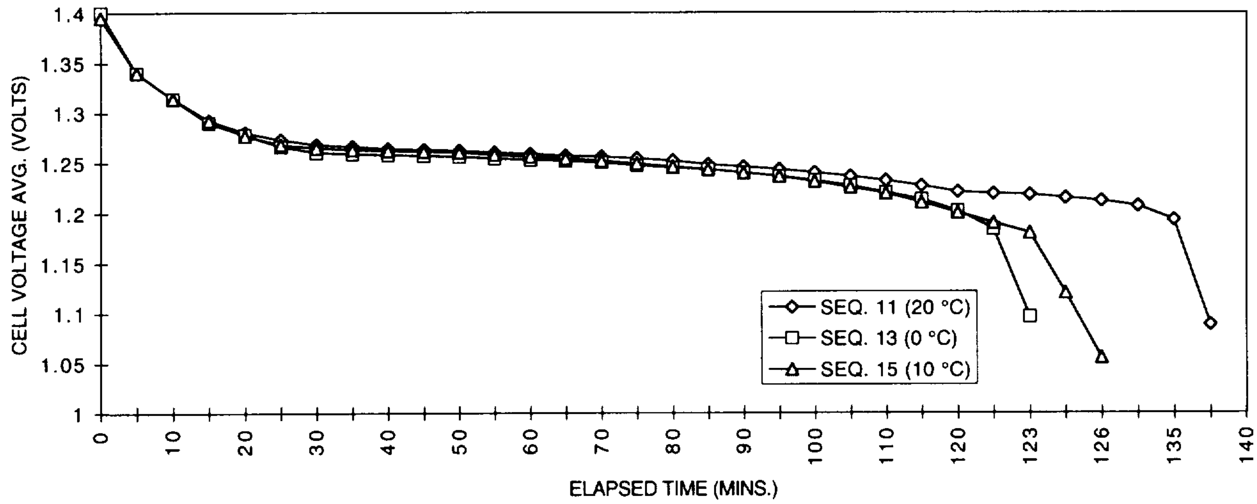


Figure 3. Test pack 3-C/2 discharge to 1.0 V/cell after charging with third electrode control.

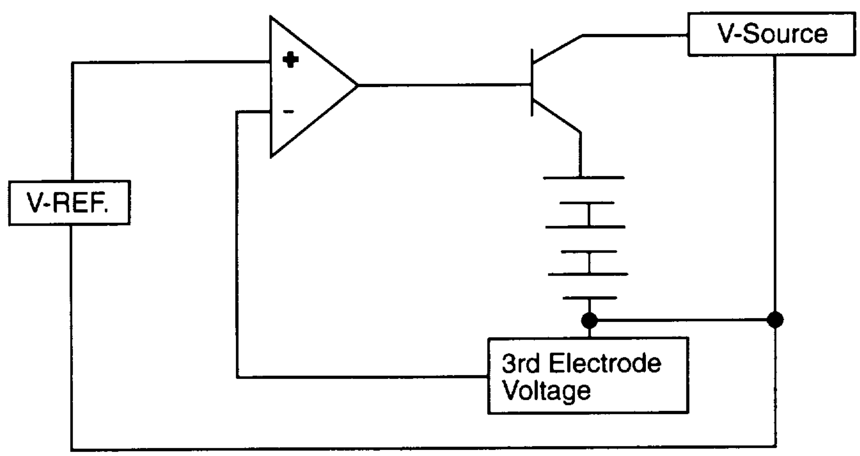


Figure 4. Simplified third electrode control circuit.

After completing 302 days of testing, Pack No. 3 was discharged to 4.5 V (0.9 V/cell). Each cell was then drained with a 1 ohm resistor for 16 hours and placed on open circuit for 24 hours. After completing 5 hours of open circuit stand, all five cells had recovered to above the specified 1.15 V per cell. The test confirmed that there were no soft shorts in these cells.

The test pack was tested for overcharge characteristics at various times during the test program. Data from two of the tests, orbit condition No. 6 and No. 107, are shown in figure 5.

Figure 6 shows the battery charge current using third electrode charge control. The charge control concept was to control the amount of charge current to keep the third electrode voltage constant rather than the battery voltage.

Pack No. 3 completed the simulation of five accelerated eclipse seasons prior to the start of the first spacecraft

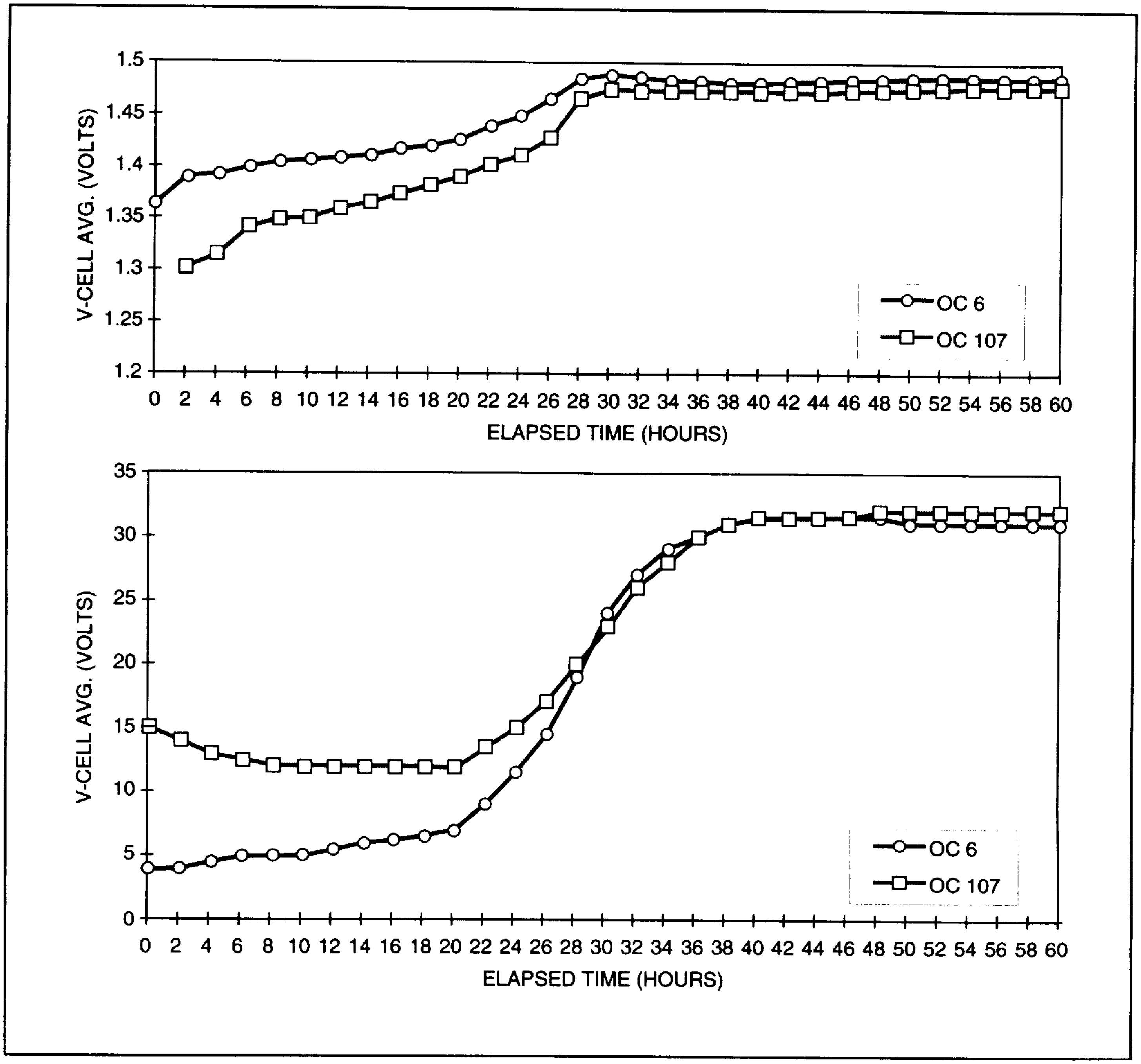


Figure 5. Pack 3: average cell voltage, and pressure, C/20 overcharge at 0 °C for 60 hours.

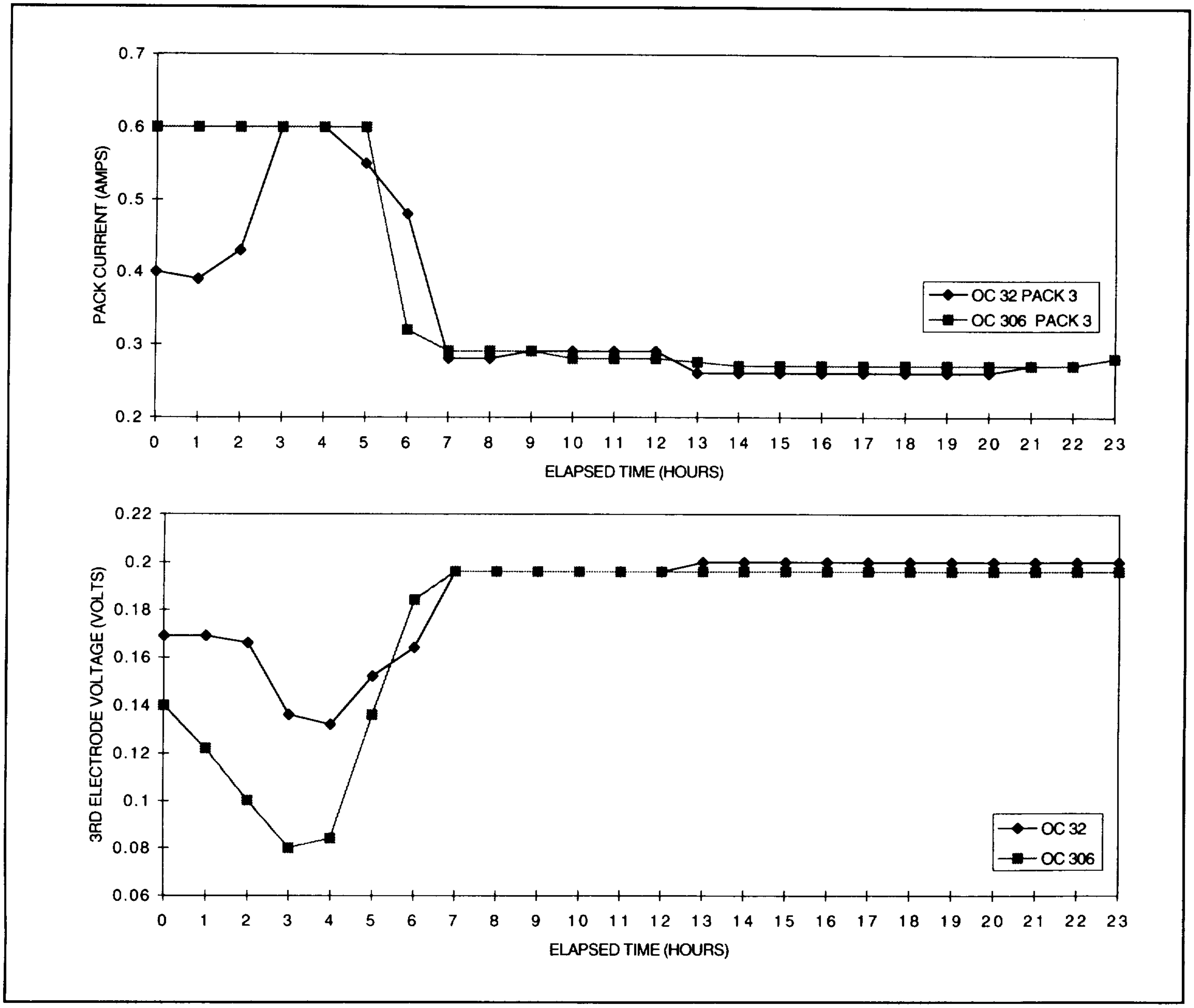


Figure 6. Pack 3: recharge current and third electrode voltage profiles using third electrode charge control at 10 °C.

eclipse season. The average cell voltage versus day-in-eclipse was plotted for eclipse seasons 1 and 4 in figure 7. The discharge currents for eclipse No. 1 and No. 5 versus day-in-eclipse are as shown in figure 8.

4.4 GSFC INCOMING CELL INSPECTION

One hundred and five 6 Ah cells purchased from GE were inspected by GSFC personnel. The following list of visual, mechanical, and electrical parameters was used for the inspection.

- Visual
  - Pits
  - Chips
  - Burrs
  - Weld uniformity
  - Cracks
  - Solder Tinning
  - Leak test,
- Mechanical
  - Cell thickness
  - Cell width
  - Cell height
  - Case height
  - Terminal dimensions
  - Weight, and
- Electrical measurements
  - Resistance - Negative to Positive terminal
  - Resistance - Negative terminal to third electrode.

One cell was rejected after failing two leak checks. Two cells exhibited minor cosmetic defects. They were slight imperfections in terminal ceramics, terminal solder flaw, and uneven weld seam. However, we considered both cells were acceptable.

4.5 CELL ACCEPTANCE

After the group of 104 cells was visually, mechanically, and electrically inspected and accepted by GSFC's incoming inspection, it was placed in cell groups and processed through an electrical acceptance test. Each group was tested in accordance with the "GSFC Test Plan Cell Acceptance Test Plan for Nickel-Cadmium Cells" — X-711-76-143. A brief description and sequential order of the test conducted are given below:

- 25 °C Conditioning,
- 20 °C Capacity,
- 20 °C Voltage Recovery,
- 10 °C Capacity,
- 0 °C Overcharge,
- 10 °C Burn-in Cycles (10)
- 10 °C Capacity,
- 20 °C Capacity, and
- Leak Test.

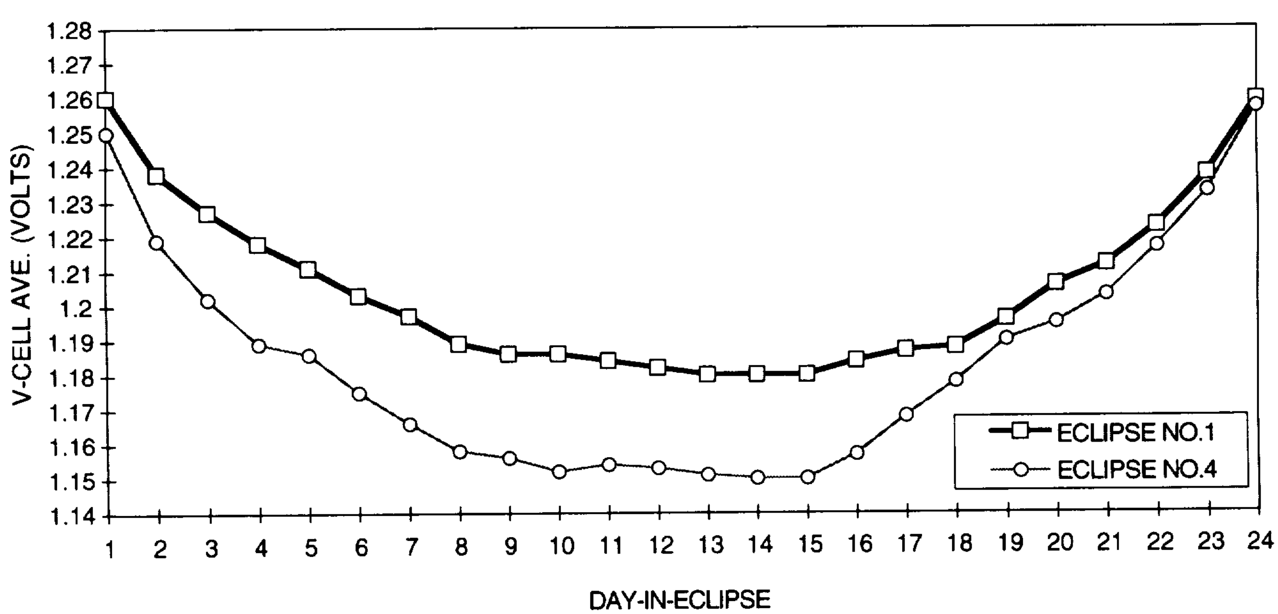


Figure 7. Pack 3: avg. peak discharge cell voltage v. day-in-eclipse at 10 °C.

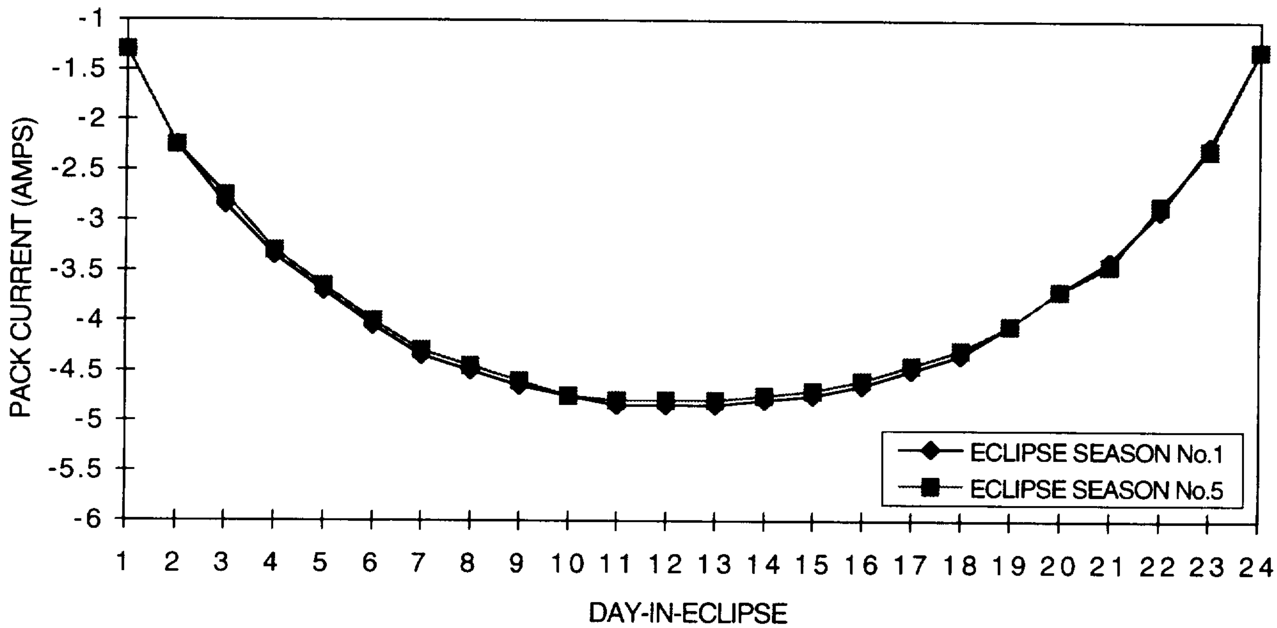


Figure 8. Pack 3: peak discharge current v. day-in-eclipse at 10 °C.

4.6 CELL PERFORMANCE REQUIREMENTS

During the acceptance test, each test parameter was evaluated to assure acceptable quality. Some tests were evaluated on a go/no-go basis in accordance with specified acceptance criteria. An example is the voltage recovery test which specifies that a cell's voltage shall recover above 1.15 V during the 24 hour open- circuit stand. A second example is the leak test where a verified leak is considered unacceptable. Minimum and maximum limits, specified in the "Acceptance Test Plan" — X-711-76-143, are described in the following paragraphs.

The minimum cell capacity limit at 20 °C is defined as 6.6 Ah (10% above the name plate capacity). The minimum capacity specified at 10 °C is 85 percent, and at 0 °C is 80 percent, respectively, of the delivered 20 °C capacity.

The maximum end-of-charge (EOC) cell voltage limit for each test temperature listed in the test plan is listed as follows:

- 1.48 V @ 25 °C,
- 1.50 V @ 20 °C,
- 1.51 V @ 10 °C, and
- 1.52 V @ 0 °C.

4.7 CELL ACCEPTANCE TEST SUMMARY

Because the cells were tested sequentially in cell groups, test data was acquired over a 4-month time period. Of the parameters EOC cell voltage and cell capacity, were critically evaluated.

The EOC cell voltages (C/20 charge for 23 hours) were recorded throughout the entire acceptance test period. Data from orbit conditions (OC) 10, 22, and 24 conducted at 0, 10, and 20 °C were plotted on histograms, and are summarized as shown in Table 1-1.

Table 1-1. Cell Acceptance Test: EOC Voltages

OC No.	Temp. (°C)	Cell Voltage (V)			
		Avg.	Hi	Lo	Spread
10	0	1.472	1.482	1.460	0.022
22	10	1.450	1.457	1.442	0.015
24	20	1.443	1.454	1.433	0.021

Table 1-2 also contains a capacity (C/2 discharge to 1.0 V/ cell) summary of the total group of cells (104) recorded from OC 11, 23, and 25 at three test temperatures.

Cells in **bold** print (Table 2) represent cells selected for one-half of Battery S/N 05 and plain print represents cells selected for the remaining half. The cells group this way provided a best capacity match in the battery. Cell 10A was the third electrode cell selected for the battery.

5.0 BATTERY DESIGN

5.1 CONCEPTUAL DESIGN

The initial IUE battery-design concept specified two 17-cell, 12 Ah, batteries. The first battery (integration battery) was fabricated and scheduled for test in accordance with the IUE subsystems specification, "Environmental Test Specifications for IUE subsystems, — IUE-320-74-008. The battery had been vibration tested and evaluated for magnetic properties prior to the second spacecraft design phase at which time a redesign was required to provide two 6 Ah batteries due to a spacecraft weight constraint. Because the 12 Ah battery frame had been proven acceptable during the vibration and magnetic subsystem test, a decision was made to design the 6 Ah battery frame in accordance with the criteria established during the initial design phase for the 12 Ah battery. Because the latter 6 Ah battery design (figure 9) was selected for flight, this report will be confined to the 6 Ah batteries. The characteristic features of the 6 Ah design are as follows:

- Seventeen (17) cells,
- Weight of seventeen cells - 4.763kg,
- Weight of finished battery - 5.749kg,
- Ratio of battery weight to cell weight - 1.21, and
- Overall battery dimensions - 23.6 cm x 13.4 cm x 13.3 cm.

Table 1-2. Cell Acceptance Test: Capacity

OC No.	Temp. (°C)	Capacity (Ah)			
		Avg.	Hi	Lo	Spread
11	0	5.960	6.400	5.500	0.900
23	10	7.133	7.600	6.500	1.100
25	20	7.395	7.800	7.000	0.800

Table 2. Cell Capacity Histogram for Flight Battery S/N 05

CAPACITY (Ah)	CELL SERIAL NUMBER									
7.6	**39**	**64**	**80**	**65**						
7.5	**44**	**51**	**68**	**86**	**91**	40	90	102	110	(20 °C)
7.4	58	75	89	*10A*						
7.3										
7.2	**39**	**44**	**51**	**86**						
7.1	**64**	**68**	**80**	**91**	28					(10 °C)
7.0	40	75	90	102	110					
6.9	**65**	89	*10A*							
6.8										
6.7										
6.6										
6.5										
6.4										
6.3										
6.2	**51**	**64**	**91**							
6.1	**44**	**86**	**80**	**65**	**68**					
6.0	58	40	75	90	89					(0 °C)
5.9	**39**	102								
5.8	110									
5.7	*10A*									

5.2 BATTERY PACKAGING

Specific criteria was established to provide guidelines to be used during the selection of parts and materials, and to establish assembly procedures to be used during battery packaging. They were illustrated in figure 9 as follows:

- Cell sides to be isolated from the battery case and adjacent cells with laminated glass sheet 0.025 cm (.010") thick,
- Cell edges to be isolated from the battery case and adjacent cells with laminated glass sheet 0.38 cm (.015") thick,
- Cell bottoms to be isolated from the battery case with a continuous sheet of kapton type H film 0.013 cm (0.005") thick,
- Cell bottoms to be lightly and uniformly coated with C6-1102 thermal grease for thermal conduction to the battery base plate,
- Laminated glass sheets 0.004" to 0.006" thickness used to obtain desired preload on end plates,
- End plates, center divider/strut, side walls and base plate made from magnesium (AZ31B-H24, OQ-M),
- Cell hold-down clamps made from aluminum alloy 6061-T6,QQ-A-250/11 with pads made of laminated phenolic sheet to electrically isolate the clamp from adjacent cell, and
- Dummy cell fabricated from aluminum honeycomb.

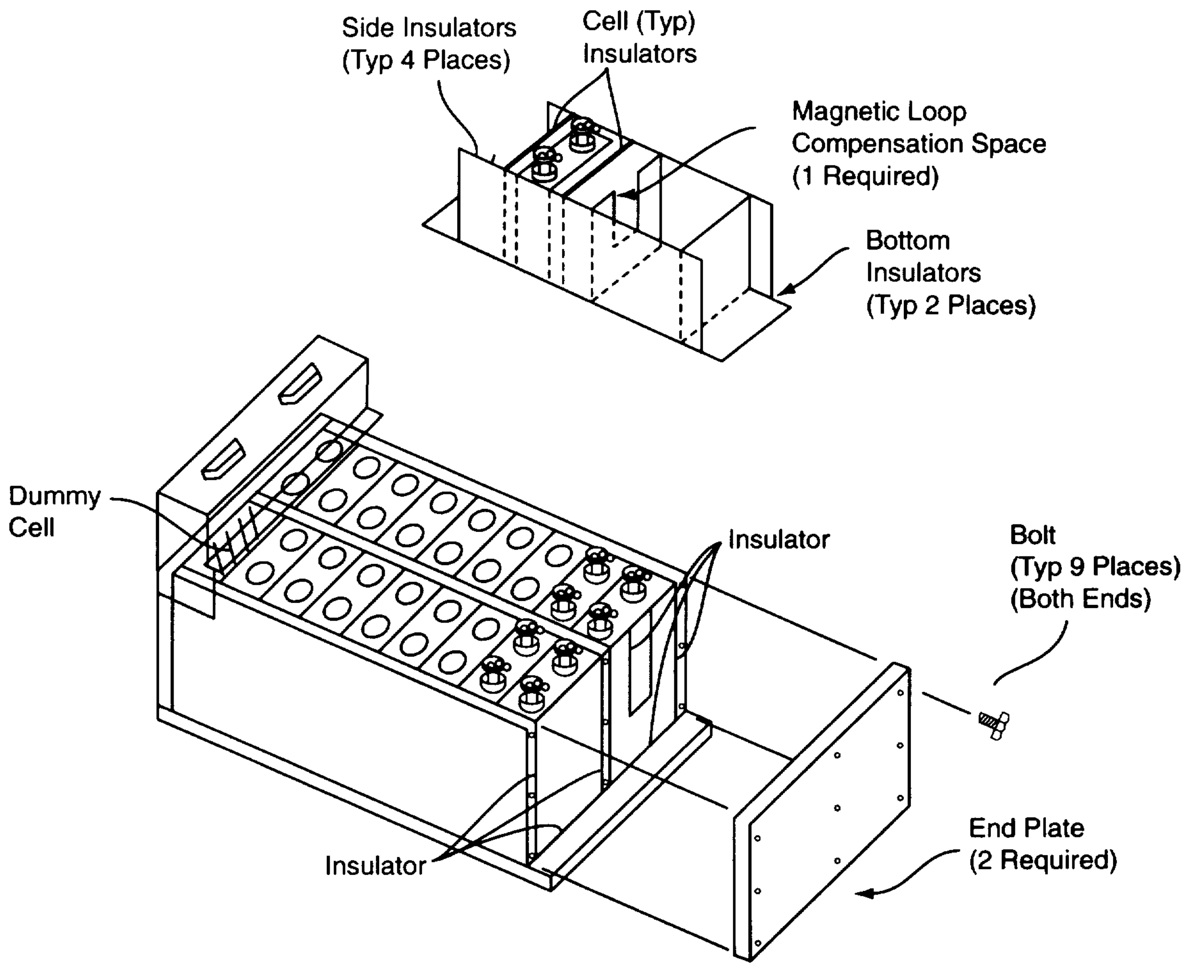


Figure 9. 6 Ah battery structure.

5.3 BATTERY ELECTRICAL INTERFACE

Each battery contains a test connector designed to provide an interface with ground support equipment (GSE). This approach provides access to each battery without interrupting normal spacecraft operations maintained through the power connector. A brief description of each connector is shown below, noting the electrical parameters available at each connector.

- Power connector with 44 pins providing the following
  - Load lines,
  - Battery voltage sense, and
  - Thermistor interface.
- Test connector with 25 pins providing the following
  - Cell voltage,
  - Battery voltage,
  - Thermistor interface Umbilical, and
  - Provision for charge/discharge of independent of power connector.

6.0 BATTERY FABRICATION

6.1 GENERAL

The IUE battery fabrication and assembly procedures are given in detail in the GSFC document "Battery Fabrication and Acceptance Test Plan for International Ultraviolet Explorer (IUE) 6 Ah Batteries," —X711-78-5. This report describes the major steps in the processes.

6.2 MECHANICAL ASSEMBLY

Each battery was mechanically assembled in stages to accommodate the application of C6-1102 thermal grease to the bottom surface of each cell followed by the frame assembly. The main sequential assembly steps used are as follows:

- The bottom of each cell was lightly sanded, cleaned, and coated with thermal grease,
- Main frame partially assembled (sides, bottom, and end plate on connector end),
- Cell insulator sheets installed into main frame,
- Cells inserted into main frame,
- End plate (foot end of battery) loosely installed,
- Cell hold-down clamps slightly tongued down sufficient to hold cells down,
- Battery pre-loaded to 45 ± 5 psi, shims were installed or removed until 45 psi could be obtained, and
- End plates torqued down to specified torque.

6.3 ELECTRICAL ASSEMBLY

Each battery was electrically assembled and wired in accordance with the steps outlined below.

- Complete battery cable harness fabricated and installed into a connector bracket,
- Wires routed and mechanically attached to termination points,
- Thermistors mounted and soldered to harness wiring,
- Magnetic compensation formed and installed,
- All harness wires soldered in place, and
- Connector bracket torqued down.

6.4 CONFORMAL COATING APPLICATION

Conformal coating was applied to the battery connectors in accordance with GSFC "Space Qualified, Non-Sag Urethane, a Pressure-Supplied Insulation for Harness Cable Connectors"—MTR-No. 313-001. Battery harness and wiring were secured to the frame using the cable ties and staked with the urethane compound.

6.5 BATTERY IDENTIFICATION

The identification of each battery was engraved on the bottom edge of the rear end plate in 1/8 inch high letters, in accordance with the IUE subsystems criteria. Each label identifies the project, system, subsystem, and serial number of each battery i.e., "IUE-P-BAT-01, 02, 03, 04, 05, and 06."

6.6 GSFC BATTERY ACCEPTANCE

The four IUE flight and flight spare batteries (S/N's 03, 04, 05, and 06) were acceptance tested and environmental tested at GSFC. Whenever possible, the batteries were tested in pairs (two flight and two spares) in accordance with two documents "Battery Fabrication and Acceptance Test Plan For International Ultraviolet Explorer (IUE) 6 Ah Batteries," — X-711-78-5 and "Environmental Test Specification for IUE Subsystems," — IUE-320-74-008, Rev. 1.

Sections of document X-711-78-5 describe the test plan used during the battery acceptance test. A brief test outline showing the sequence of test events is as follows:

- 25 °C Conditioning,
- Magnetic Test,
- 10 °C Low rate battery charge (simulated spacecraft charge),
- 10 °C Burn-In cycles (5),
- 10 °C Capacity,
- 20 °C Voltage Recovery,
- Vibration,
- Leak Check,
- Peak Load Discharge,
- Thermal-vacuum,
- 0 °C Capacity,
- 20 °C Capacity,
- 10 °C Capacity, and
- Leak Check.

6.7 MAGNETIC TEST

After three of the batteries were conditioned and recharged, they were tested in accordance with the IUE test document, IUE-320-74-008, Rev. 1 for magnetic characteristics at the GSFC Magnetic Test Range. The data acquired are presented in Table 3.

The data shown indicates that all the batteries tested had magnetic properties well below the specified limit (less than 100 gauss) when tested at three discharge current levels during the stray field test.

6.8 VIBRATION

Table 4 lists the specified test conditions for the spacecraft subsystems in the IUE document "Environmental Test Specifications for IUE Subsystems" — IUE-320-74-008, Rev. 1. During the test the battery voltage, current, and third electrode signal were monitored. After the test data was analyzed and each battery visually inspected for structural damage, batteries proceeded through the acceptance testing.

Immediately following the vibration test, each battery was completely discharged and each cell was checked for leaks with a phenolphthalein solution in accordance with the test procedure in the GSFC document "Battery Fabrication and Acceptance Test Plan for International Ultraviolet Explorer (IUE) 6 Ah Batteries," — X-711.78-5.

After being discharged at a C/2 rate (3 A) for 70 minutes, each battery was subjected to a 3C (18 A) discharge load for 5 minutes. The battery was considered acceptable when its voltage was above the 17 V during the 3C discharge period.

Table 3. IUE Batteries Magnetic Test Characteristics

Bat. S/N	Initial Perm	Post 15 Gauss Exp	Post De-Perm	Stray Field (Gauss)								
				X	Y	Z	X	Y	Z	X	Y	Z
	(Battery Current)				(1.5A)			(3.0A)			(6.0A)	
03	110.5	1980	24.8	16.1	14.3	5.4	23.2	26.9	10.1	38.4	49.7	18.6
04	110.0	1780	20.7	6.4	17.1	6.3	16.5	28.6	14.8	40.4	50.1	31.5
06	69.4	604	8.5	5.6	10.9	4.0	9.8	23.0	9.2	18.3	42.3	41.5
Field Magnitude in Gamma at 12"												

Table 4. IUE Battery Acceptance Vibration Test Levels

SINUSOIDAL (4 OCT./MIN.)	FREQUENCY (Hz)	LEVEL (0 to peak)
- THRUST AXIS	5-7.2	0.75" dA*
	7,2-15	2.0 G
	15-21	7.0 G
	21-35	4.5 G
	35-60	4.0 G
	100-200	2.0 G
- LATERAL AXIS	5-25	2.5 G
	25-45	3.5 G
	45-200	1.5 G
RANDOM	FREQUENCY (Hz)	
- THRUST AND LATERAL AXIS	20-150	0.032 G^2/Hz
	150-250	+6dB/OCT.
	200-1000	0.083
	1000-2000	-3dB/OCT. G^2/Hz
SHOCK	FREQUENCY (Hz)	ACCELERATION
- THRUST AND LATERAL AXIS	200-500	+10.5dB/OCT.
	500-4000	300 G

* double amplitude displacement

6.9 THERMAL VACUUM

Figures 10 and 11 show the thermal vacuum test data for the Batteries S/N 03 and S/N 04, respectively. These plots are typical representations of the third electrode charge characteristics at the extreme ends of the temperature test range from -10 to 30 °C. Figure 10 indicates that battery S/N 03 was slow in reaching the level of temperature stability acquired by the other Battery S/N 04. A comparison of the temperature profile curves shown in figures 10 and 11 indicates that the temperature delta levels existed during the first 120 minutes of test. After the test was terminated, the surface of both batteries was examined for anomalies. It was noted that battery S/N 03 contained a void in the layer of thermal grease. Measurements were made of the bottom surface of the four batteries (S/N's 03 to 06) to check the mating surface flatness. (See figure 12). It can be noted that the surface of Battery S/N 03 contained a 0.011" curvature. This area was the position of the thermal grease void and was noted for future corrective action.

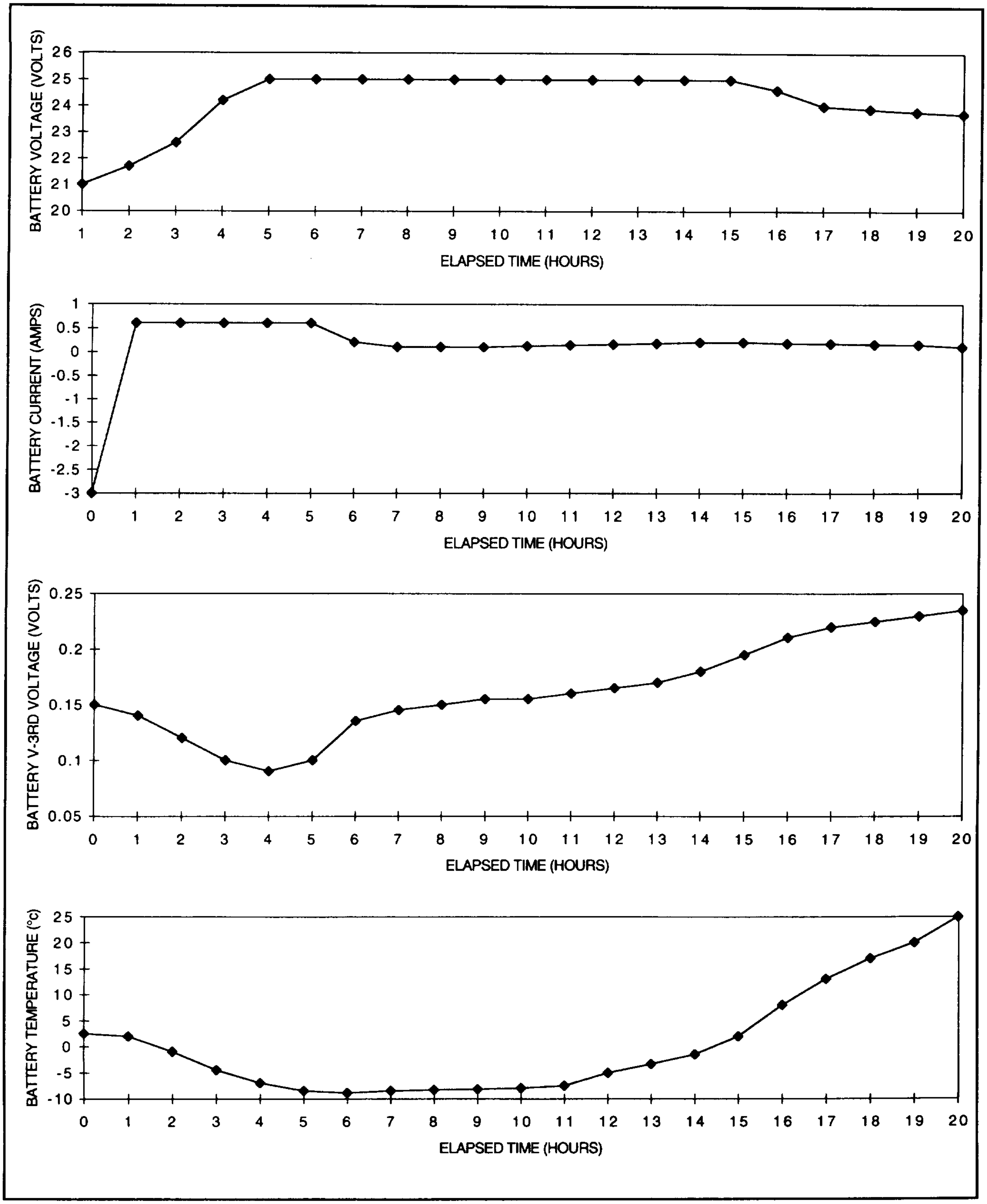


Figure 10.
Battery S/N 03: "voltage," "current," "third electrode voltage," and "temperature" v. elapsed time - subsystem thermal vacuum.

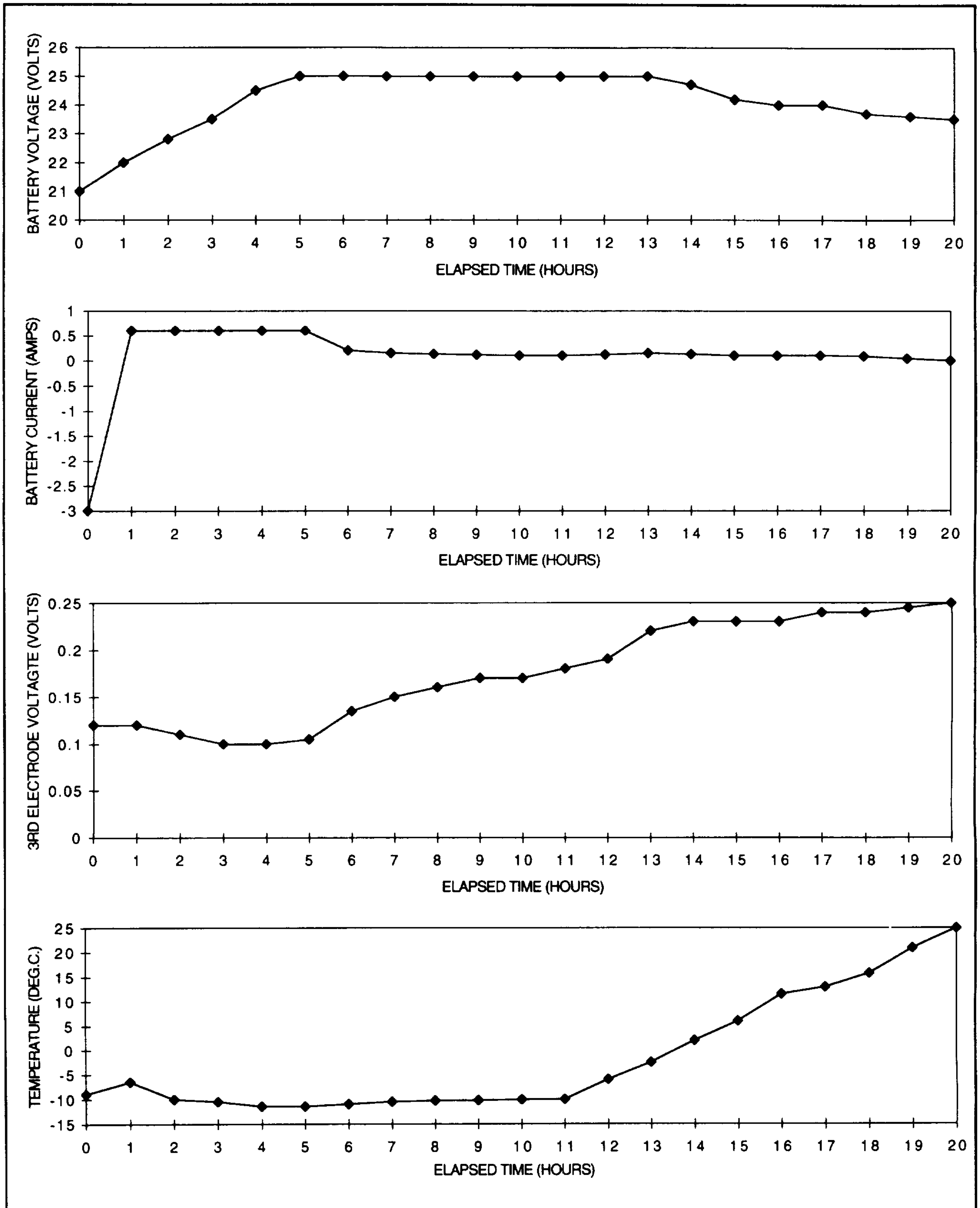
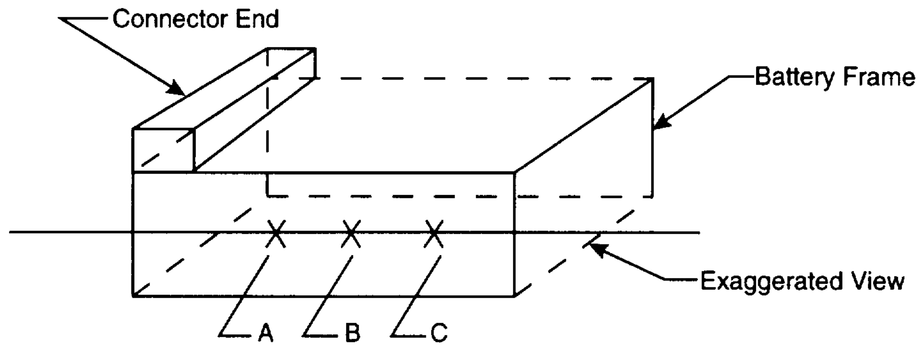


Figure 11.

Battery S/N 04: "voltage", "current", "third electrode voltage", and "temperature" v. elapsed time - subsystem thermal vacuum.



Battery Serial No.	Flatness Deviation (Inches)		
	A	B	C
03	.002	.006	.011
04	.004	.007	.005
05	.003	.008	.008
06	.004	.006	.006

Figure 12. 6 Ah battery: mating surface flatness.

6.10 CAPACITY

EOC cell voltages (see Table 5) acquired during battery capacity tests were analyzed for conformance and uniform frequency distribution to assure that the battery was below the acceptance levels specified in the battery acceptance test document X-771-85-5. The data is also used as background for future reference. Table 6 is a summary of capacity data at 0, 10, and 20 °C.

Table 5. Battery EOC Cell Voltages

BatteryNo.	Temp. (°C)	Cell Voltage (V) Avg.	High	Low	Spread
S/N 03	0	1.452	1.458	1.448	0.010
	10	1.440	1.448	1.440	0.008
	20	1.422	1.425	1.421	0.004
S/N 04	0	1.441	1.448	1.438	0.010
	10	1.436	1.441	1.434	0.007
	20	1.417	1.420	1.415	0.005
S/N 05	0	1.448	1.451	1.445	0.006
	10	1.442	1.445	1.432	0.008
	20	1.421	1.422	1.419	0.003
S/N 06	0	1.469	1.475	1.464	0.011
	10	1.442	1.447	1.438	0.009
	20	1.414	1.417	1.412	0.005

7.0 SPACECRAFT PERFORMANCE ENVIRONMENTAL TEST

7.1 THERMAL VACUUM

When the batteries were recharged with the GSE prior to the final spacecraft thermal vacuum test, the data indicated that there was an anomaly associated with the third electrode signal level or control circuitry to the inside battery (S/N 03). The data indicated that the third electrode signal level may have degraded approximately 50 percent from the initial acceptance test baseline value. It was also indicated that the signal was approximately 15 percent lower in amplitude than the level of the other Battery (S/N 04). However, data throughout the remaining time of the thermal-vacuum test indicated that both batteries appeared to be operating as designed at the specified third electrode levels.

Table 6. 6 Ah Battery Capacity Summary

Type Capacity	48 Hour Condition	Post Cycle	20 °C	Post VIB Peak load	Post T.V.	20 °C	0 °C
Sequence No.	4	12	14	18	26	28	31
Test Temp. (°C)	20	10	20	20	0	20	0
Capacity (Ah)			BATTERY S/N				
7.40						03	
7.35			03,04,05				
7.30			06			04	
7.25							
7.20							
7.15							
7.10						05	
7.05	05						
7.00	06	04,05				06	03
6.95	03						04
6.90	04	03					
6.85							
6.80							
6.75		06					
6.70					03		05,06
6.65							
6.60							
6.55							
6.50					05,06		
6.45							
6.40				04			
6.35				03,05,06	04		
6.30							
6.25							
6.20							
Temp. °C			Recorded Battery Test Temperatures				
Batt SN-03	24.6	12.6	22.3	22.2	7.4	22.3	4.9
Batt SN-04	23.8	12.9	21.1	20.6	4.4	21.1	2.6
Batt SN-05	26.5	12.7	24.7	24.5	5.0	23.2	0.5

7.2 THERMAL BALANCE

During an overcharge test with power dissipations of 8.1 and 12.3 W, a temperature delta of 5 °C was noted between Battery S/N 03 and Battery S/N 04. This condition was attributed to the abnormal decrease of the third electrode signal level of Battery S/N 03, which allowed excessive charge current, without the designed current taper. Figure 13 presents data acquired during the test where it can be noted that there was a 9 °C temperature delta between the inside battery and the spacecraft main platform. However, when the batteries were maintained in a low rate trickle charge (0.13 A) the data indicated a 3 °C temperature delta between the batteries and a 4 °C delta between battery S/N 03 and the spacecraft main platform. During the 30 °C thermal balance test that the main spacecraft platform temperature was equal to the temperature of the inside battery but was 2 °C hotter than the outside battery, with the batteries in an open circuit condition. This reversal of temperature indicated that the outside battery was acting as a heat sink for the spacecraft main platform.

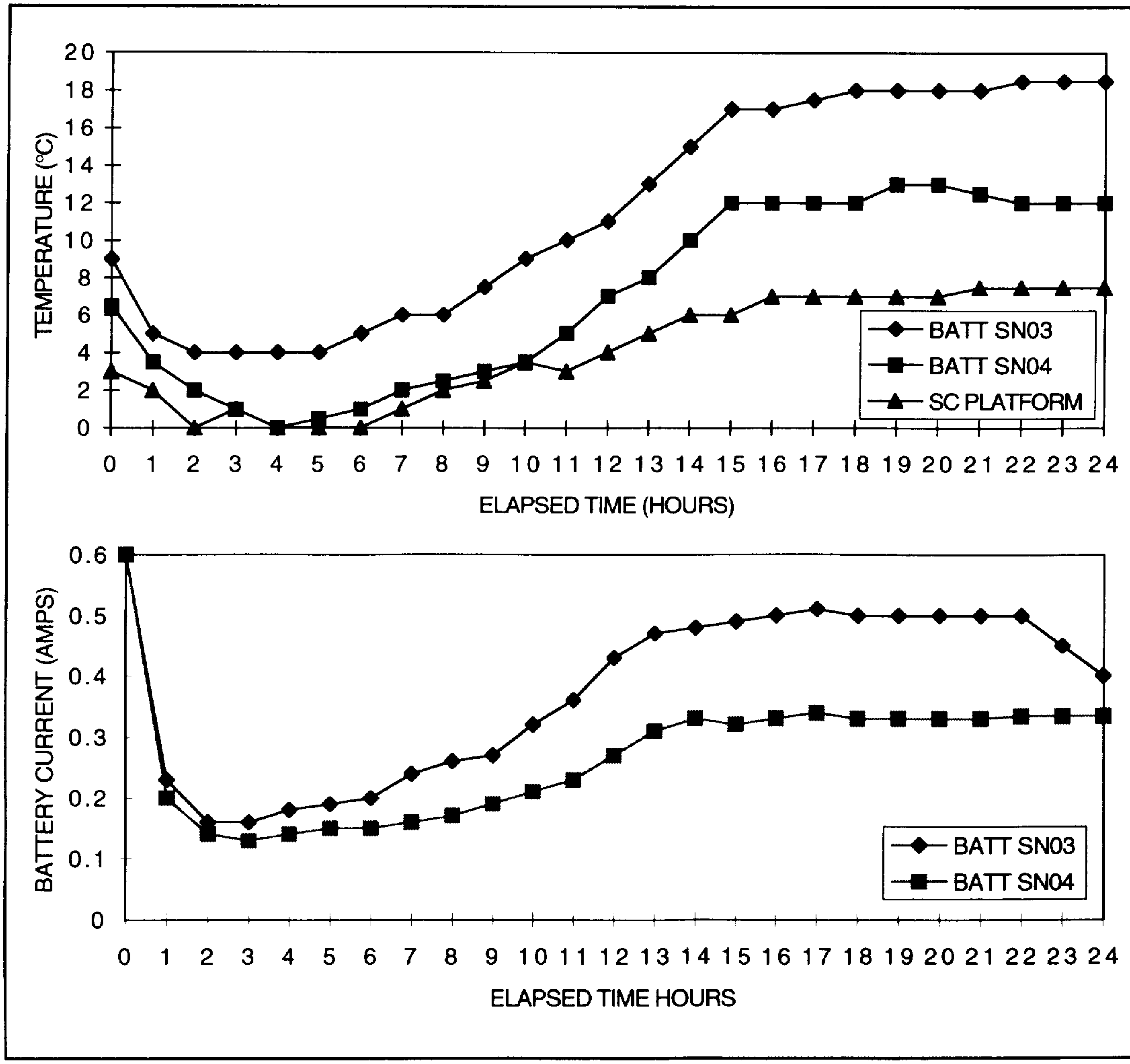


Figure 13. Batteries "temperature," and "current" v. elapsed time - spacecraft thermal balance.

8.0 SPACE FLIGHT PERFORMANCE

8. 1 PRELAUNCH PREPARATIONS

As previously mentioned in the introduction section of this document, flight Batteries S/N's 03 and 04 were replaced with flight spare Batteries SN's 05 and 06 during prelaunch preparations. Spacecraft telemetry identified Battery SN 05 as Battery No. 1 and Battery S/N 06 as Battery No. 2. The remaining sections of this document will refer to these batteries as No. 1 and No. 2.

Approximately 10 hours prior to launch, the batteries' main chargers were turned on to bring the batteries to a full state of charge. Figure 14 is a plot of the charge characteristics during the prelaunch, launch, and post launch events. It can be noted from the curve that the full state of charge was obtained after approximately 2 1/2 hours of charge, during the launch preparations, at which time the spacecraft umbilical power was reduced to maintain the batteries at a low rate trickle charge (0.15 A).

8. 2 LAUNCH

The spacecraft was launched with the batteries' main chargers turned on to insure adequate recharge capability during the critical stages of the launch, transfer and park orbits. Figure 14 also illustrates the battery characteristics observed during this time period.

8.3 IN ORBIT

Once the mission orbit was obtained, both batteries were switched to the low trickle charge mode to maintain the batteries in a full charge state and storage mode of operations until required for the first eclipse season. Immediately after the deployment of the solar array panels the spacecraft telemetry indicated that there was a 8 °C temperature delta between the batteries. Figure 15 is a plot of the battery temperature during the first 46 days of life in orbit. A temperature delta of this magnitude had not been previously observed during any of the environmental tests or in this mode of operation conducted at the spacecraft level. Table 7

Table 7. Battery Temperature Gradient Under Various Conditions

Type Environment	Platform Temp.	Power Dissipation *1 Inboard	Outboard	Battery Temperature Inboard	Outboard	Delta Temperature
	(°C)	(W)	(W)	(°C)	(°C)	(°C)
Spacecraft	23	3.5	3.5	23.9	23.5	0.4
Thermal vacuum	-3	1.0	3.3	-1.7	0.0	1.7
Spacecraft	31	0.0	0.0	31.0	29.0	2.0
Thermal		2.9	2.7	10.8	6.	4.7
Balance	7	12.3	8.1	18.0	12.0	6.0
KSC**	26 *2	2.9	2.9	26.5	26.5	0.0
S/C Checkout	20 *3	3.1	4.3	22.0	22.0	0.0
In Orbit	22	2.9	2.7	20.0	12.0	8.0
		7.7	7.1	27.0	19.0	8.0

*1 Calculated from battery voltage and current
*2 Approximate temperature available of non operative units.
*3 Spacecraft on Launch Pad.

**Kennedy Space Center

presents temperature delta observed throughout the spacecraft thermal test, prelaunch checkout and space flight in the synchronous orbit. The highlights are as follows:

1. Temperature deltas were minimal during the spacecraft thermal vacuum test and KSC checkout.

2. The outboard battery was 2 °C cooler than the spacecraft main equipment platform.

3. Maximum temperature delta of 4.7 °C noted between batteries (battery dissipation was balanced during spacecraft thermal balance test.)

4. Temperature delta of only 6 °C was noted with batteries at a high (power) dissipatin of 12.3 and 8.1 W during the spacecraft thermal balance test.

5. Battery temperature delta of 8 °C was noted for the first time with the batteries in two separate power balanced conditions of 2.9-2.7 and 7.7-7.1 W, after the spacecraft achieved synchronous orbit with the solar array panels deployed.

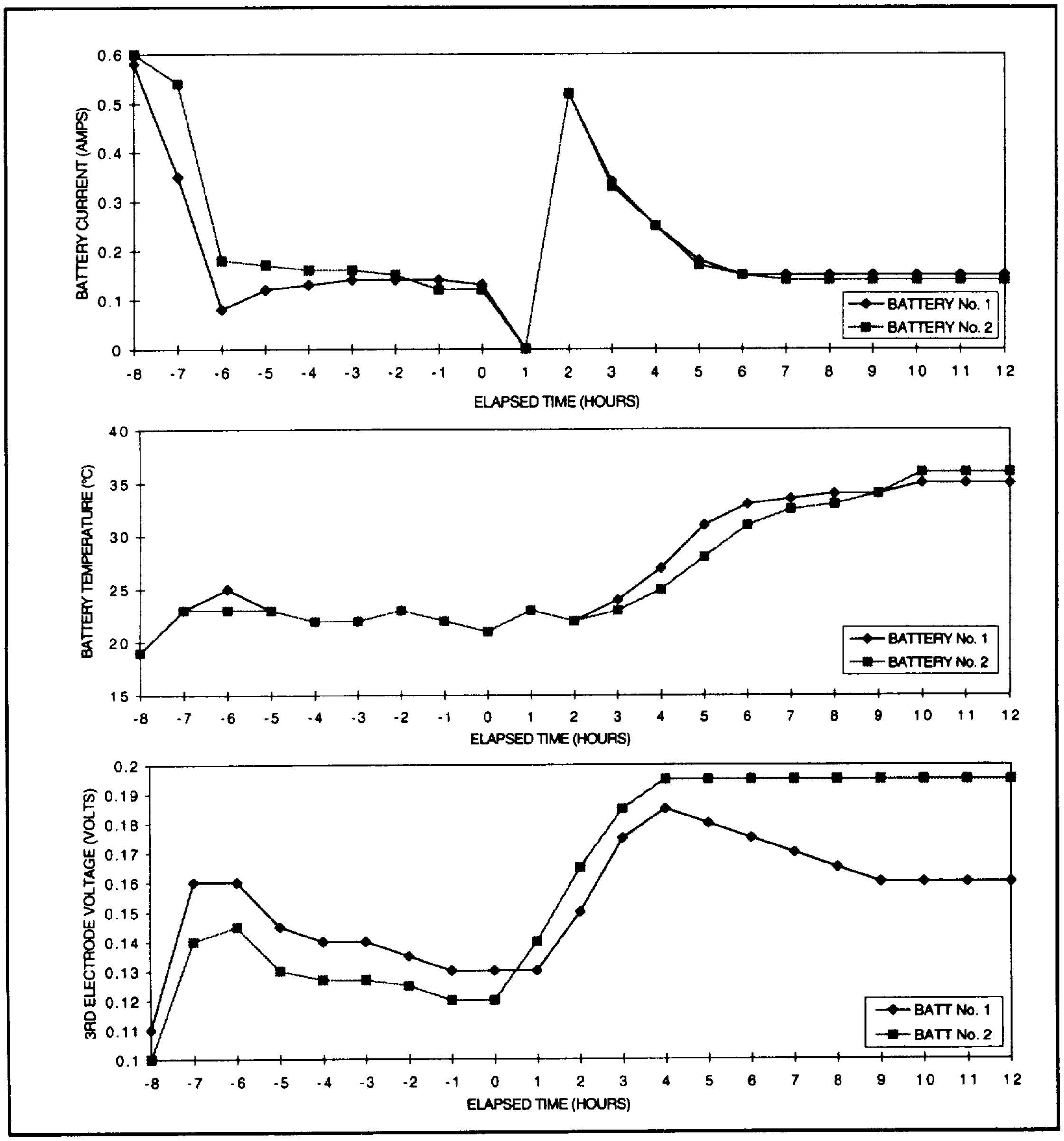


Figure 14.
Batteries "current," "temperature" and "third electrode voltage" v. elapsed time - spacecraft prelaunch/launch events.

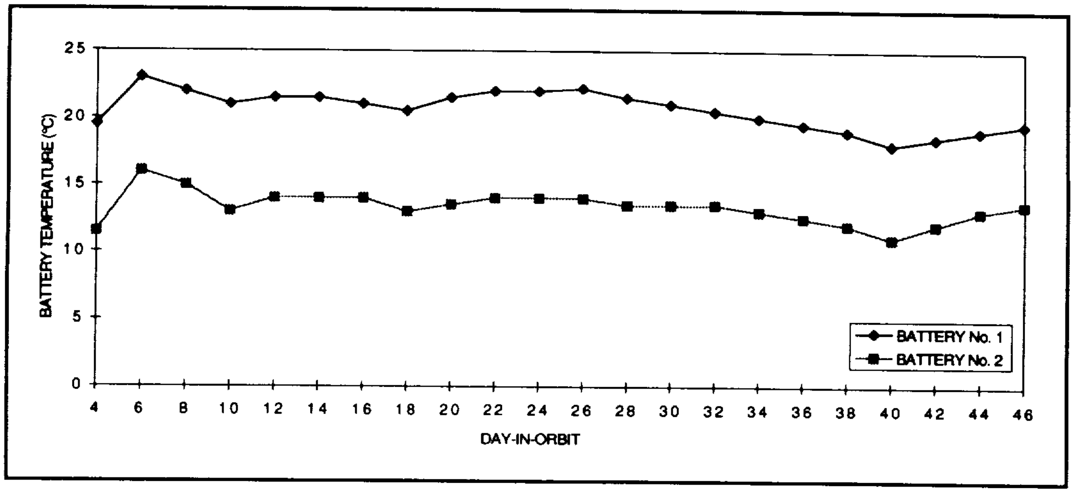


Figure 15. Batteries temperature v. day-in-orbit — spacecraft batteries on trickle charge.

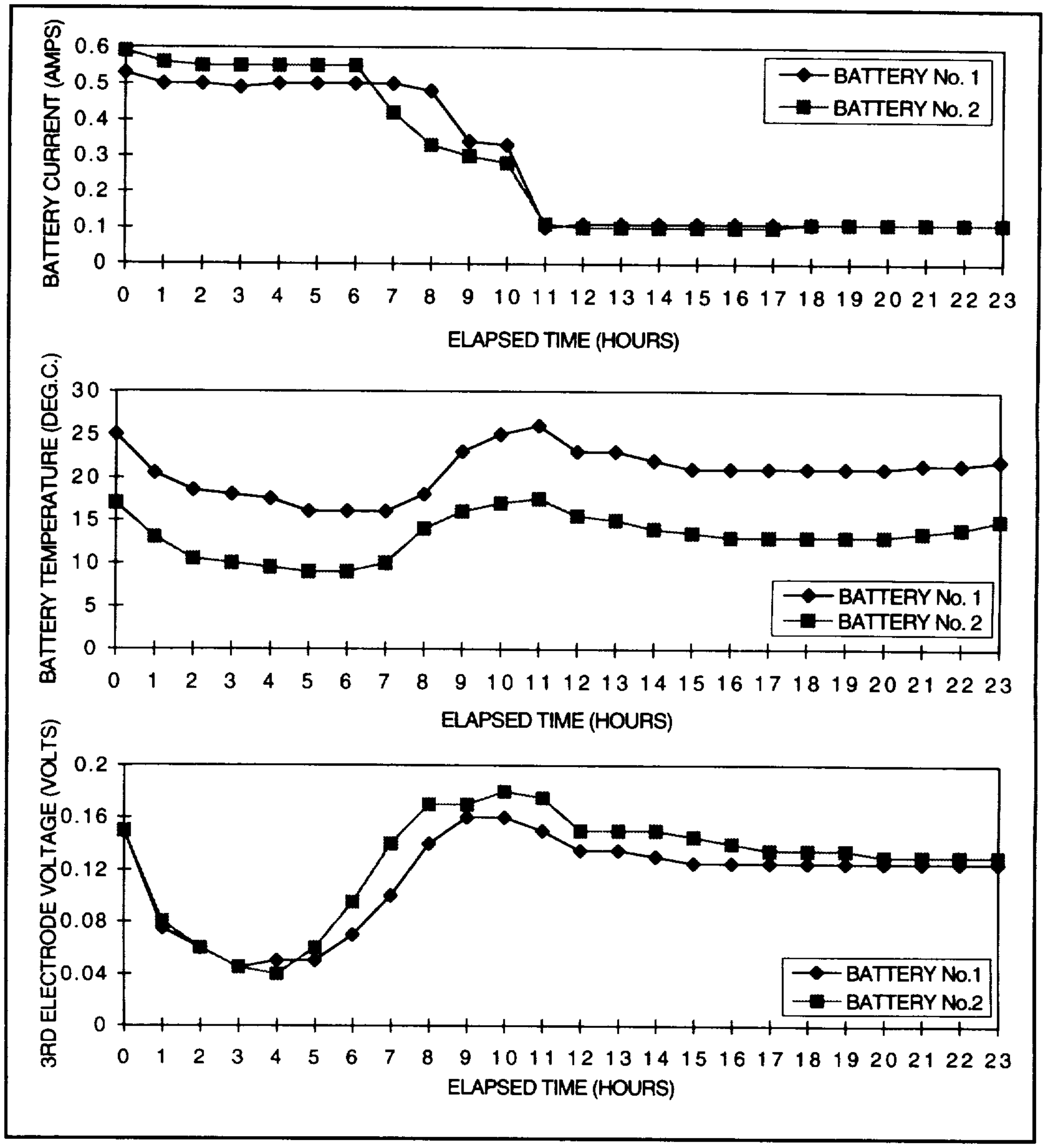


Figure 16. Spacecraft batteries recharge "current," "temperature," and "third electrode voltage" v. elapsed time - eclipse season no. 1 (after 54-minute shadow).

8.4 SPACECRAFT ECLIPSE SEASON NO. 1

Prior to the start of the first eclipse season, a review was held by the IUE project office to discuss the spacecraft battery eclipse operations. It was decided that the operational constraints selected during the initial spacecraft design phase were correct. Directives were issued by the project which again specified that the maximum battery DOD was to be 80 percent, with a maximum current limit of 4 A per battery during the eclipse periods. Directives were also issued to curtail specific space functions by turning off specified subsystems to lower the spacecraft power requirements to 164 W and maximum power limit during the eclipse. The data acquired for evaluation during the first eclipse season was used to prepare battery management guidelines during the subsequent spacecraft eclipse seasons.

Figure 16 shows the battery recharge characteristics after a 54-minute eclipse period (day 6) of the first eclipse season. Both batteries were placed on trickle charge when both of the third electrode signals were above 0.15 V and the temperature of Battery S/N 05 had reached 25 °C. When the spacecraft was positioned above 127° beta angle, the array power was reduced by approximately 50 percent. The PSE charge current was also reduced. When the spacecraft was at that attitude the batteries operated cooler and remained on the main chargers without either battery reaching the 25 °C temperature limit. Figure 17 shows the battery discharge characteristics, as observed during the 13th day of the first eclipse season during a peak period. The battery DOD was 72 percent. Even with a temperature delta of 8 °C between the batteries load sharing was excellent.

8.5 BATTERIES OPERATIONAL DIRECTIVES

Data plotted (figure 18) at the conclusion of two years of mission indicated that battery end-of-discharge (EOD). peak voltage was 19.65 V which was a substantial decrease from the 20.2 V at the peak eclipse period of eclipse season No. 1. This decrease appeared to be directly related to DOD and aging effects. Consequently, the nonessential spacecraft loads were turned off on each eclipse day when battery discharge reached 50 percent in order to extend the mission life. DOD ranges for the mission life are shown in Table 8 below.

Table 8. Spacecraft DOD Ranges

Eclipse Seasons	DOD
1-4	65.2 - 76.7
5-18	51.2 - 69.1
19-24	49.0 - 61.8
25-38	37.0 - 53.07

An initial directive specified that the spacecraft be positioned to a beta angle sufficient to allow a minimum of 8 hours of charge time prior to an eclipse period, and that the batteries could not be discharged within 2 hours of an umbra eclipse period. Data acquired from launch to August 1983 showed that the EOD voltage of Battery No. 1 was dropping faster and steeper than that of Battery No. 2 and could possibly accelerate to the undervoltage limit. The consensus was that it would be better to lose a battery than allow an undervoltage situation to deprive the spacecraft of power during an umbra period and possibly loose spacecraft control. Therefore, on August 31, 1983, a decision was made to turn the battery undervoltage circuits off.

Both batteries' third electrode circuits appeared to decrease in sensitivity in orbit and were overcharging the batteries when the battery charge taper current remained too high (0.3 A). The directive was modified to specify that the batteries be switched to a low trickle 3 hours after the start of current taper. In 1990, a decision was made to turn the third electrode control circuits off. On February 22, 1990, the circuit of Battery No. 1 was turned off. On March 19, 1990, the circuit of Battery No. 2 was turned off. Ironically, the third electrode circuit for Battery No. 1 operated according to design for one Eclipse Season. There was no plausible explanation of the behavior of the circuit for that period. The battery management was modified to specify that the main battery chargers be used to recharge the batteries during the eclipse periods to a 25 V clamp and switched to trickle charge when either battery's temperature increased by 1 °C above the minimum temperature reached during recharge.

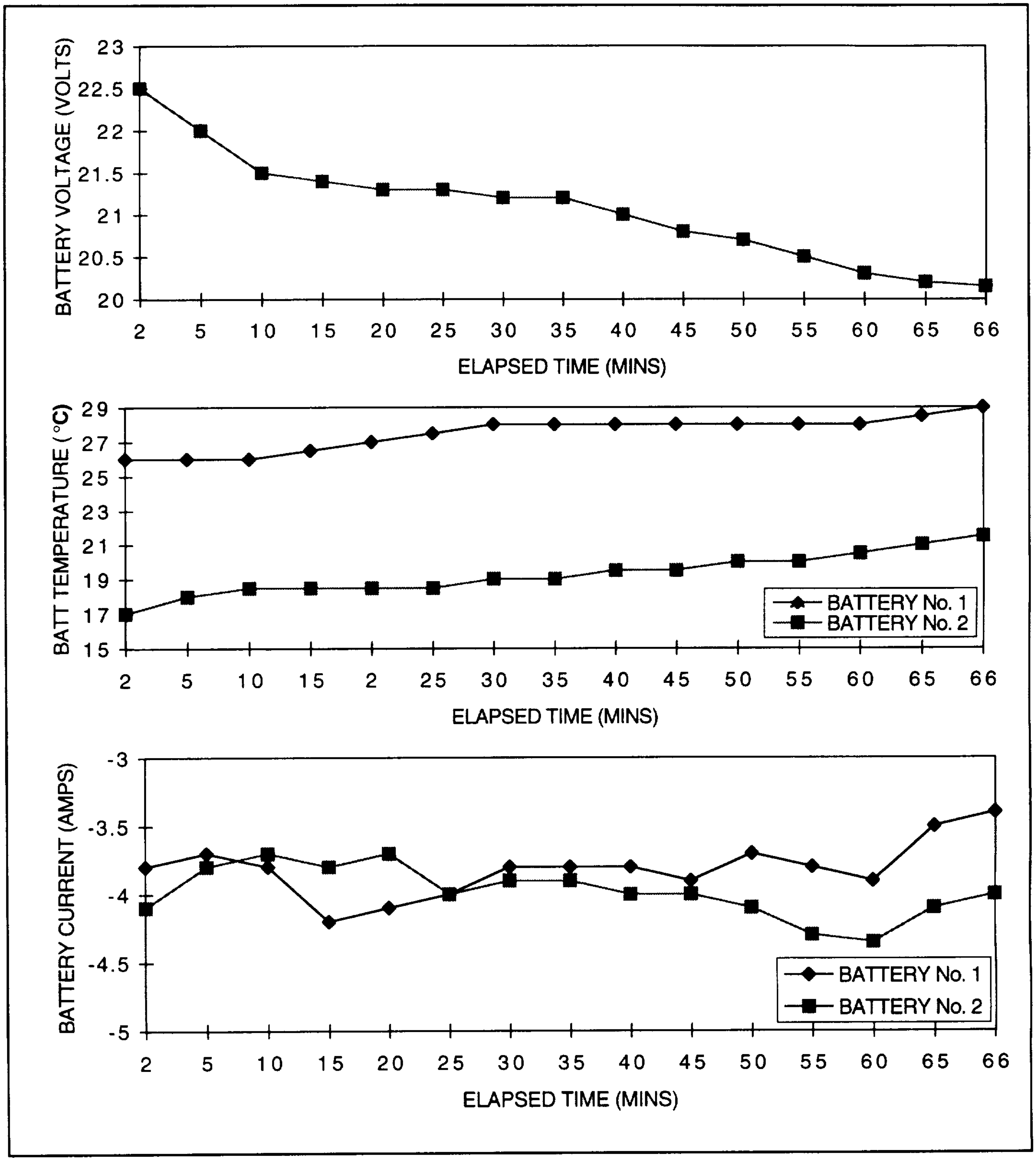


Figure 17.
Spacecraft batteries, discharge "voltage," "temperature," and "current" v. elapsed time - eclipse season no. 1 (66-minute shadow).

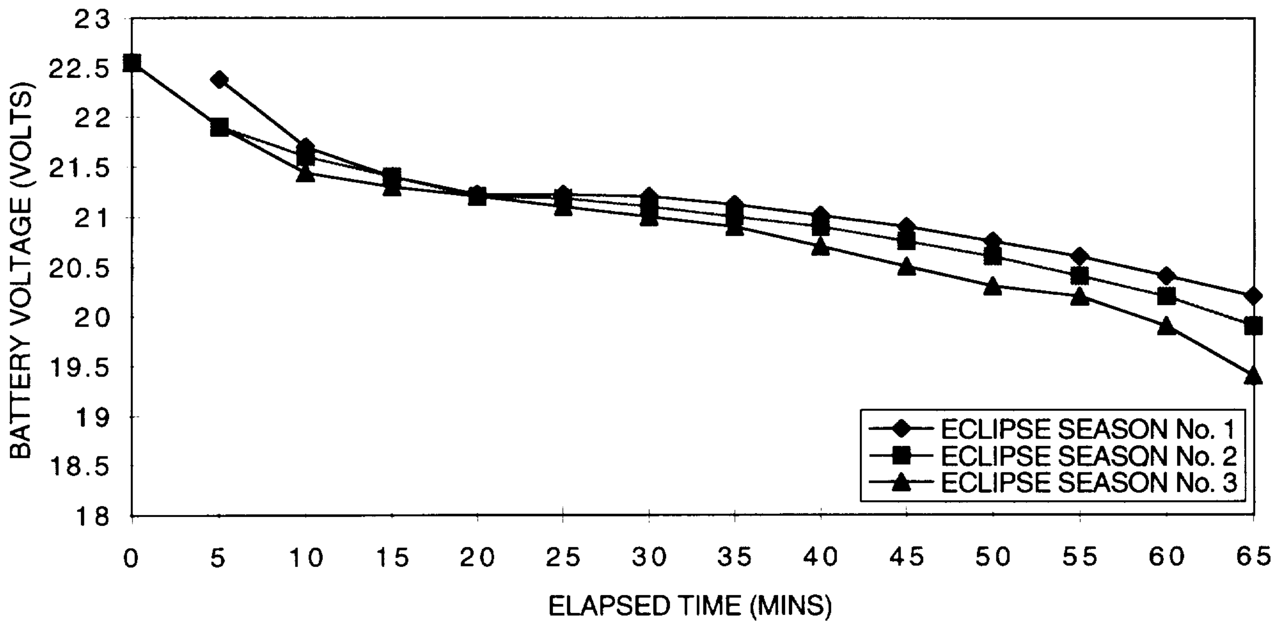


Figure 18. Spacecraft battery discharge voltage at peak eclipse period v. elapsed time.

8.6 BATTERY-IN-ORBIT OPERATIONS

During the mission life of the spacecraft (eclipse seasons 1 to 38) battery load sharing was nominal. Figure 19 illustrates the load sharing. Also shown in the illustration is that Battery No. 2 was the predominant power provider for eclipse seasons 1 to 9, and 25 to 38, and that Battery No. 1 was the predominant power provider for eclipse seasons 10 to 24. It is generally accepted that this behavior was caused by the 8 °C temperature delta between the batteries. Figure 20 illustrates the battery DOD voltage during the peak of each eclipse season. It should be noted that the lowest voltage shown may not be at the middle of the eclipse seasons because selected spacecraft systems were switched on or off as required for the mission at different times during the eclipse seasons.

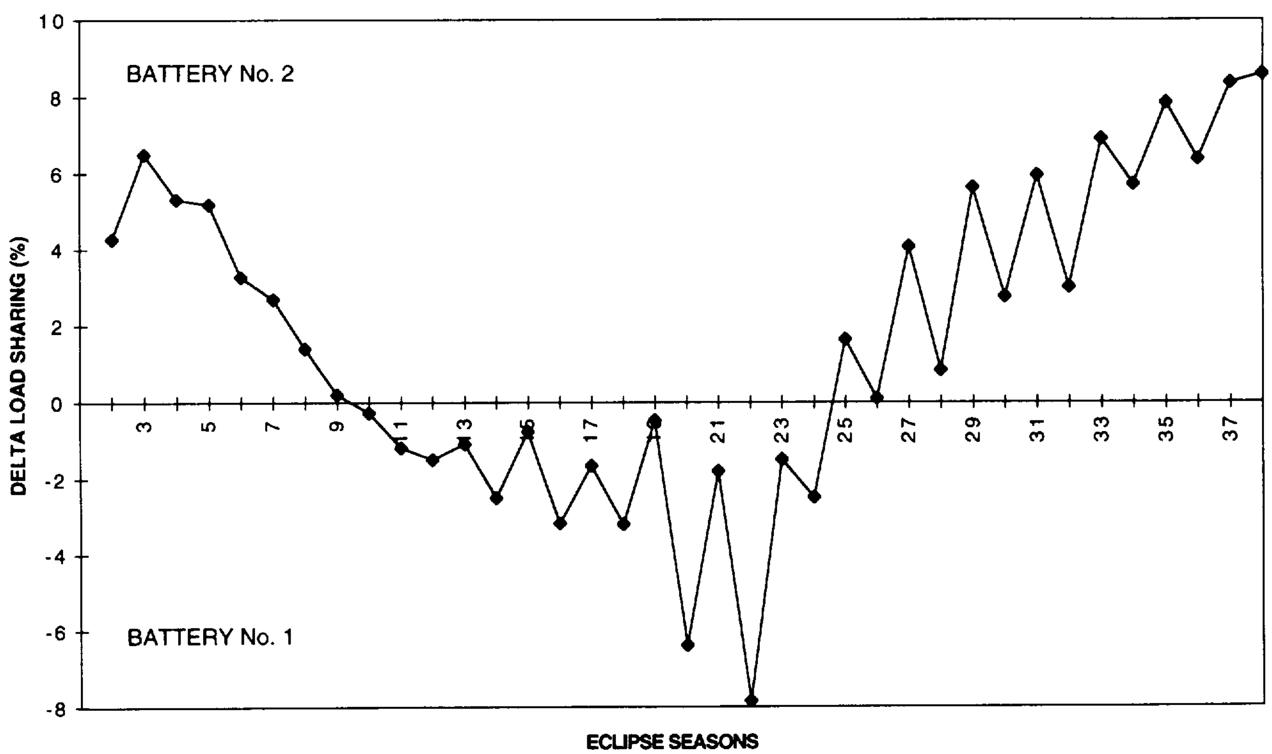


Figure 19. Batteries load sharing v. peak eclipse season.

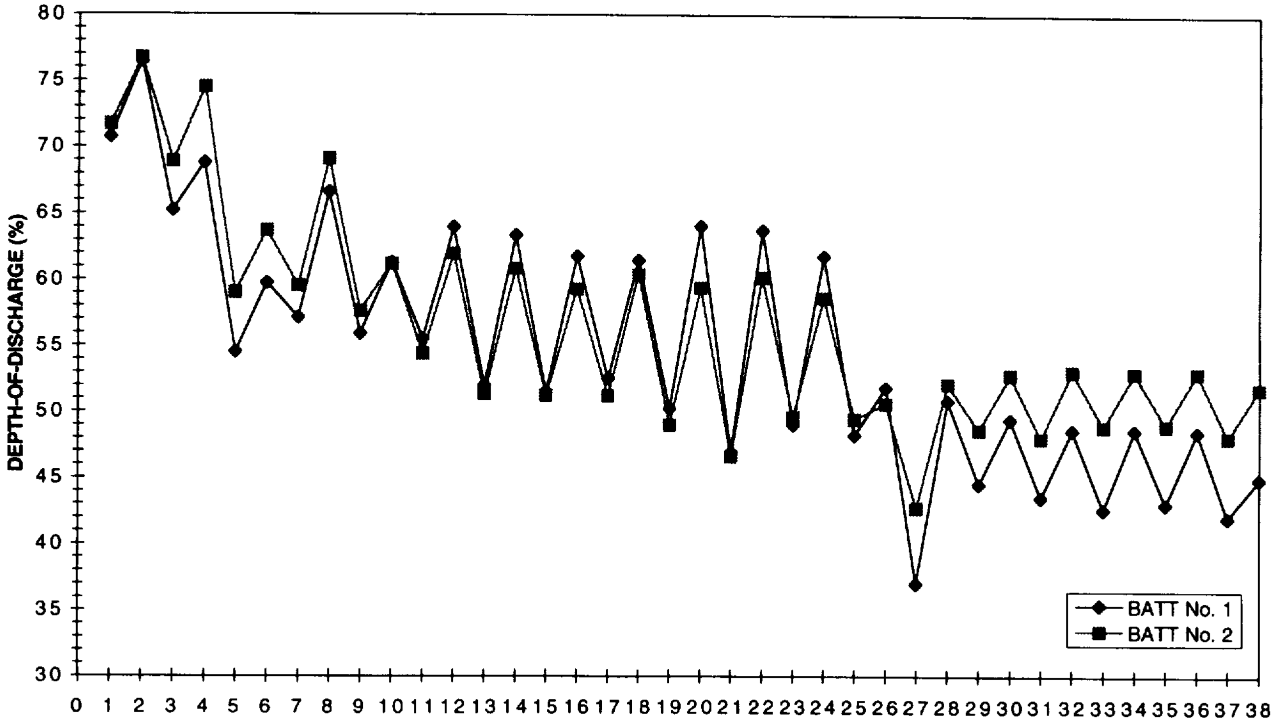


Figure 20. Batteries maximum DOD v. shadow season.

9.0 END-OF-SPACECRAFT-LIFE OPERATIONS

On September 26, 1996 the final battery tests were started in conjunction with the IUE spacecraft "End-of-Life-Operations" conducted between September 26 and 30, 1996. These final tests were conducted using conservative guidelines so as not to compromise the integrity of the spacecraft systems' final "End-of-Life Operations."

9. 1 BATTERY ON/OFF SWITCH CHECK

On day 270 (September 26, 1996) each fully charged battery was commanded off/on line individually with the spacecraft positioned to a power positive attitude to verify that the battery on/off relay control circuits were operational after being dormant for 18 years. Real-time data indicated that both batteries returned on-line after being switched off. Recorded data of the battery's third electrode measured 0.100 and 0.180 V, respectively, for Battery No. 1 and Battery No. 2 after approximately 17 minutes of

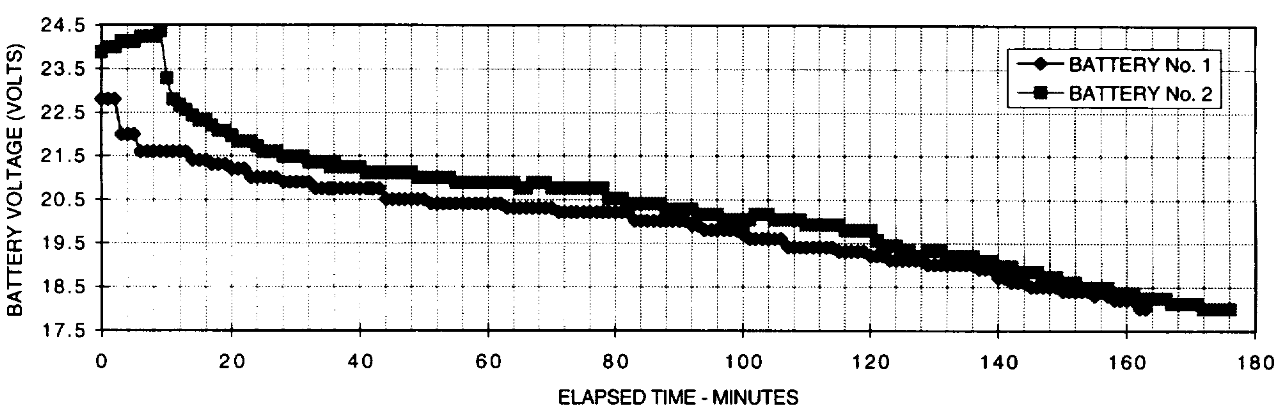


Figure 21. Batteries discharge plots (day 270).

elapsed time. The delta of 0.08 V between the battery's third electrodes verified that the third electrode of Battery No. 1 had lost sensitivity.

9.2 BATTERY DISCHARGE TO 18 V

In the second test Battery No. 1 was switched off and Battery No. 2 was discharged to a DOD of 50 percent at a 2 A rate, followed by a 1.0 A discharge until the battery reached 18 V. When Battery No. 2 reached 18 V, Battery No. 1 was switched on before removing any spacecraft loads to provide a backup power source to the spacecraft solar arrays. Both batteries were commanded to be recharged prior to discharging Battery No. 1. After the recharge, Battery No. 1 was discharged using the procedure used to discharge Battery No. 2. It should be mentioned that Battery No. 2 was discharged first since previous data indicated that Battery No. 2 was the stronger of the two batteries. Figure 21 shows discharge plots to 18 V.

9.3 THIRD ELECTRODE SIGNAL MEASUREMENT

On day 272 (September 28, 1996) a test was conducted to measure the third electrode signal level during a 1 A discharge for 1 hour. The test was started with both

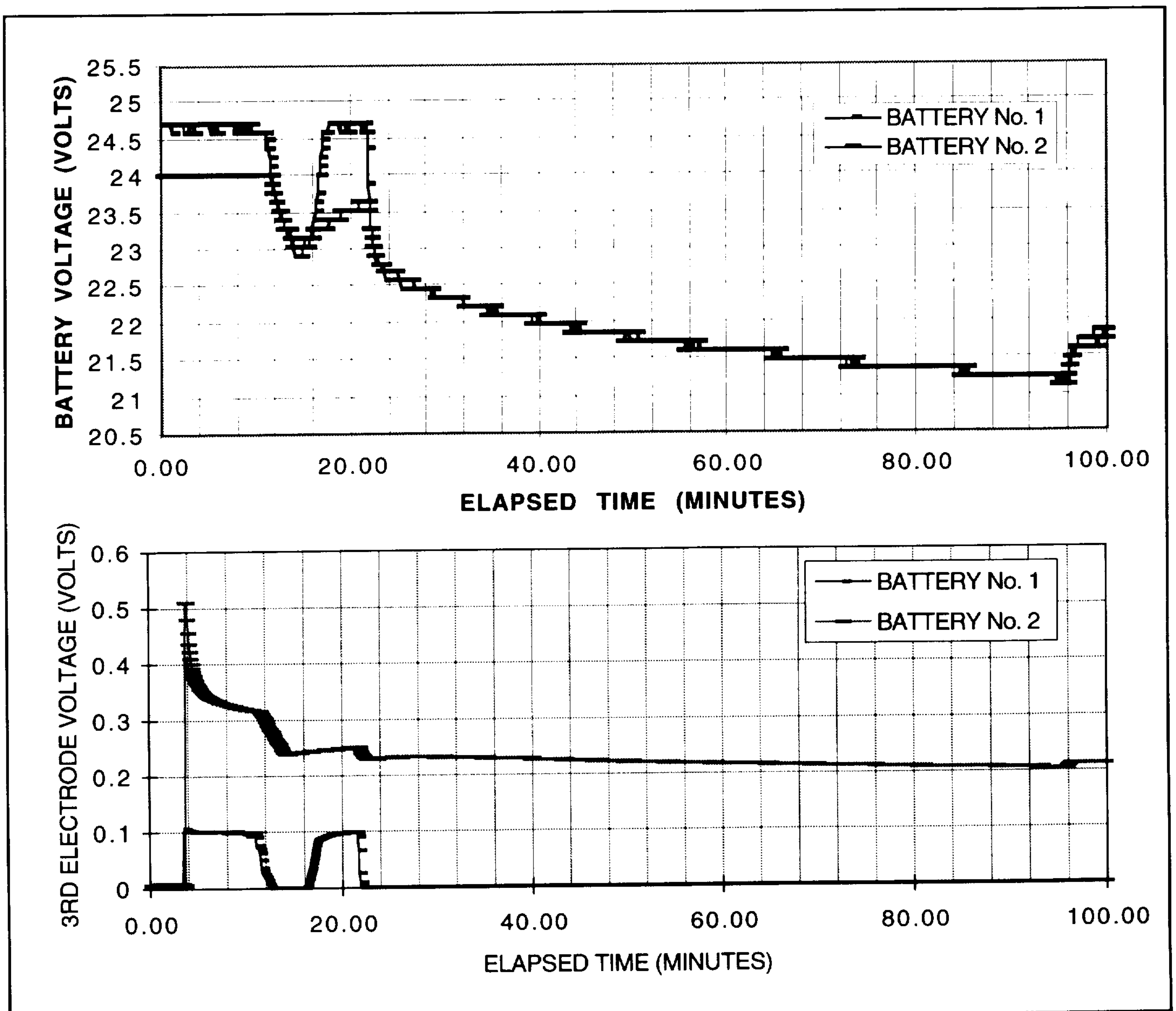


Figure 22. Batteries discharge voltage and third electrode characteristics (day 272).

batteries fully charged with each battery's third electrode circuits switched on (see figure 22). The data indicates that the third electrode voltage of Battery No. 1 is 0.0 V after 1 hour of discharge, whereas the electrode voltage of Battery No. 2 is 0.23 V.

9.4 BATTERY RECHARGE

During recharge tests at a typical C/10 (0.6 A) rate, Batteries No.1 and No. 2 voltages were gradually increasing with charging time (about 330 minutes) towards 25 V as shown in figure 23. At 25 V (24.6 V in telemetry - figure 23), the clamp voltage, the charge current tapers as shown for Battery No. 2 in figure 23. When the current reached 0.3 A Battery No. 2 was manually switched from main charger to the low trickle charger (0.1 A). It is important to note that the Battery No. 1 charge current did not taper because the test was terminated before the battery voltage reached the voltage clamp of 25 V. During the last few years of operation, Battery No. 1 was maintained on the main charger, where charge current was minimum, with weekly top-offs using the trickle charger.

Two noteworthy battery recharge characteristics are shown in figure 22. The comparison of voltages in Battery No. 1 and 2 at the start of "load discharge" was 23.6 and 24.7 V respectively. The third electrode signals read 0.0 V for Battery No. 1 and 0.230 V for Battery No. 2, indicating that the third electrode signal level for Battery No. 1 virtually lost all signal amplitude.

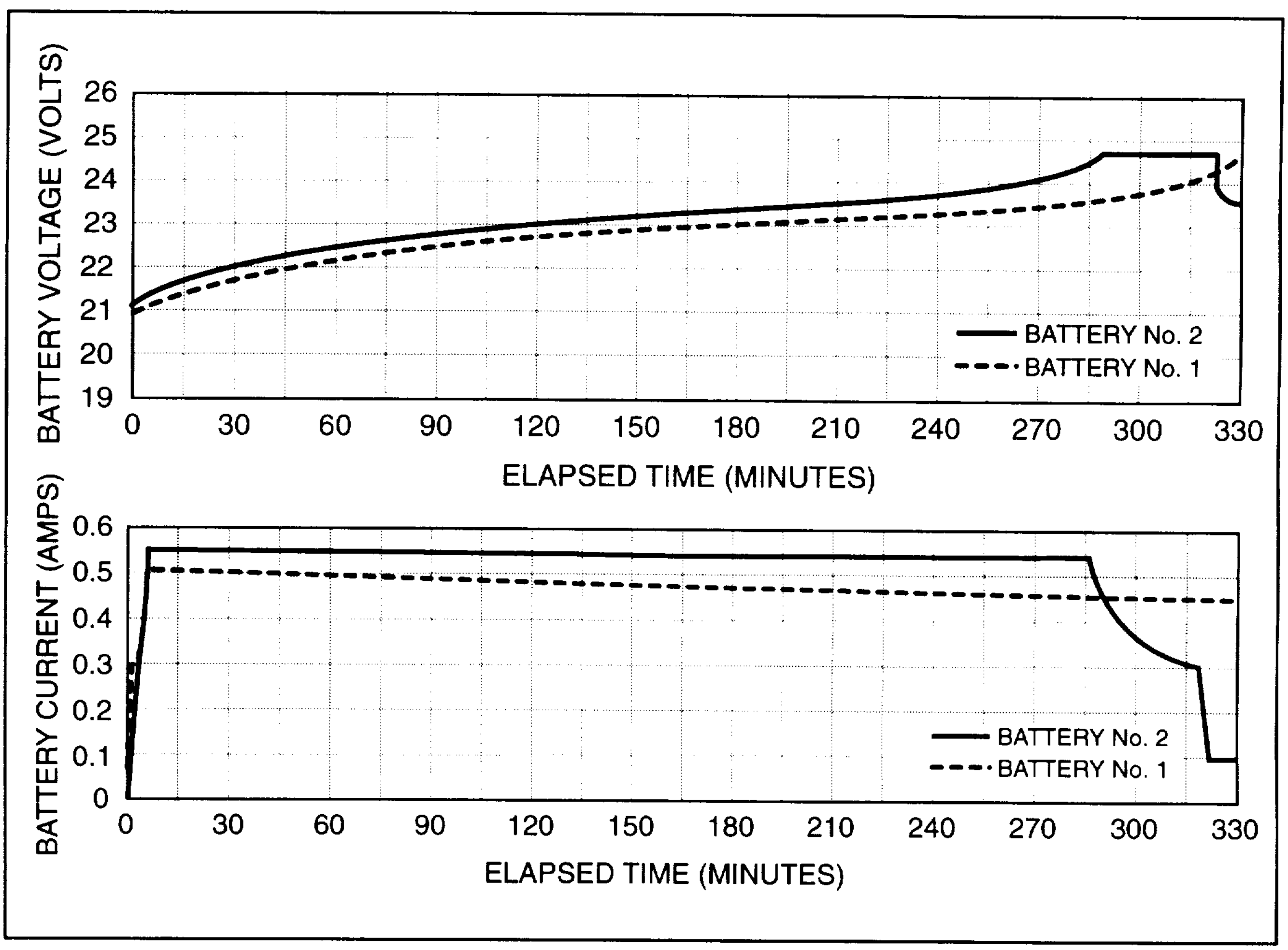


Figure 23. Batteries recharge voltage and current profiles (day 272).

9.5 BATTERY IMPEDANCE

A special test was conducted to provide data to be used for impedance measurements of each battery. The sequence of events for each battery was: (1) switch to main charger, stabilize; (2) switch to low trickle, stabilize; (3) switch to high trickle stabilize; and (4) switch to main charger. Data was acquired (figures 24 and 25) during these events and used to calculate battery impedance using the equation delta voltage/delta current = impedance i.e.

Battery No. 1: (21.96 - 21.84)/(0.536 - 0.176) = 0.33 ohms

and

Battery No. (2: 23.88 - 23.76)/(0.544 - 0.104) = 0.27 ohms

where delta battery voltage was measured during the step from low trickle to high trickle and delta current was battery current measured during the same step.

9.6 FINAL BATTERY DISCHARGE

The final test for the batteries was to discharge both batteries as deeply as possible without losing the spacecraft main bus regulator. Battery discharge occurred during the Hydrazine dump and continued until the spacecraft bus started losing regulation with the voltage of Battery No. at 16.3 V and Battery No. 2 at 16.7 V. Figures 26 and 27 show the battery voltages during this final discharge. It was important to keep the main bus in regulation and active to the support final battery and spacecraft turn off operations.

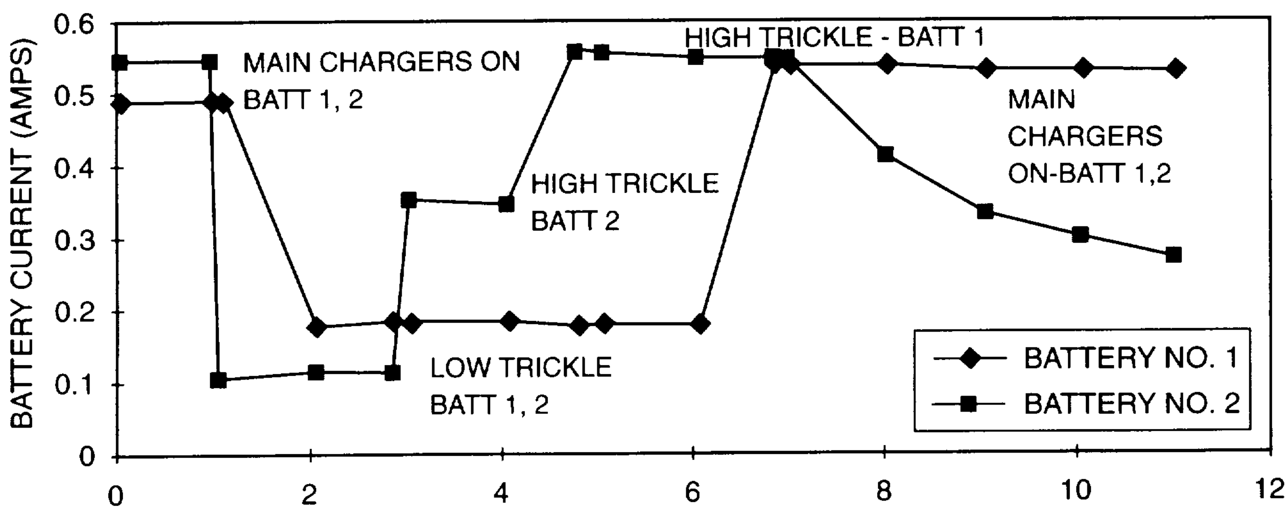


Figure 24. Batteries current - impedance data.

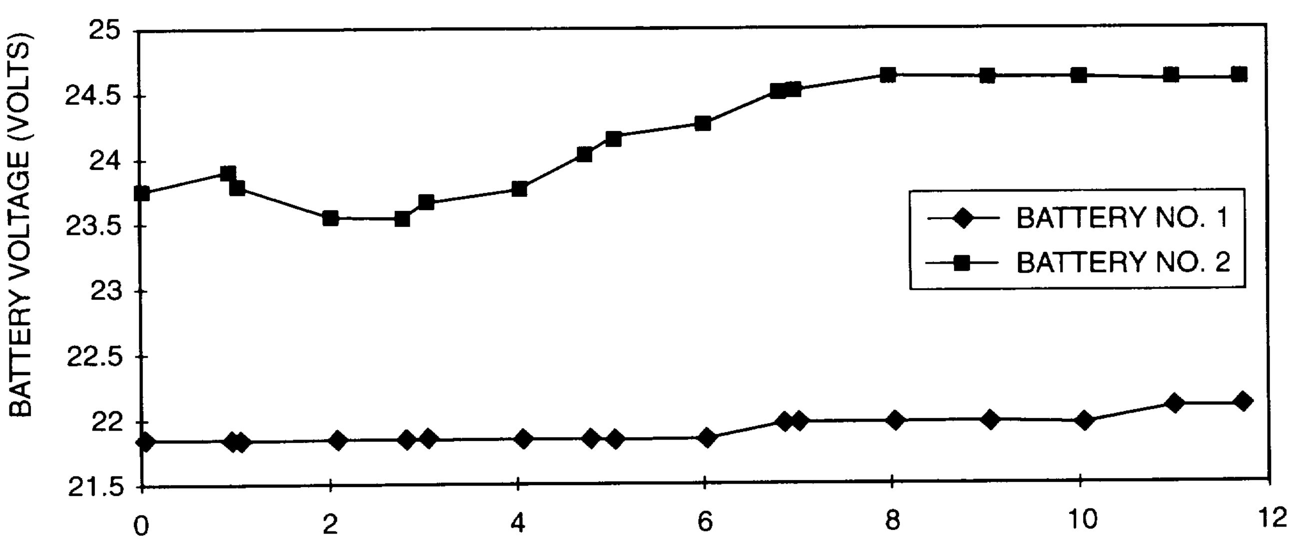


Figure 25. Batteries voltage - impedance data.

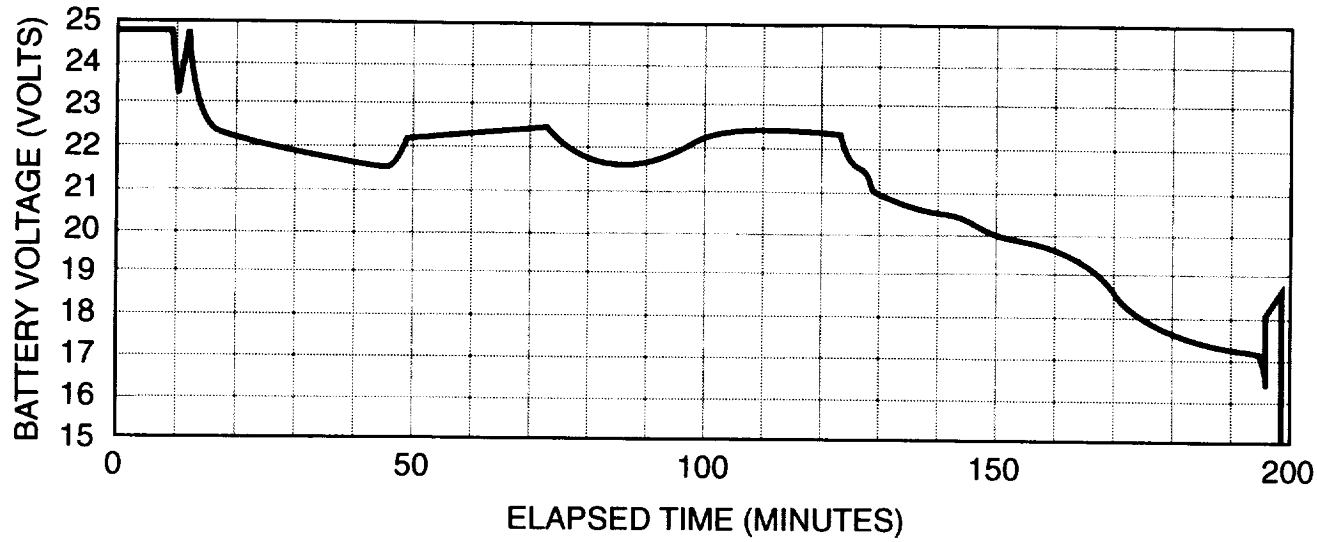


Figure 26. Battery No. 1 discharge (day 274).

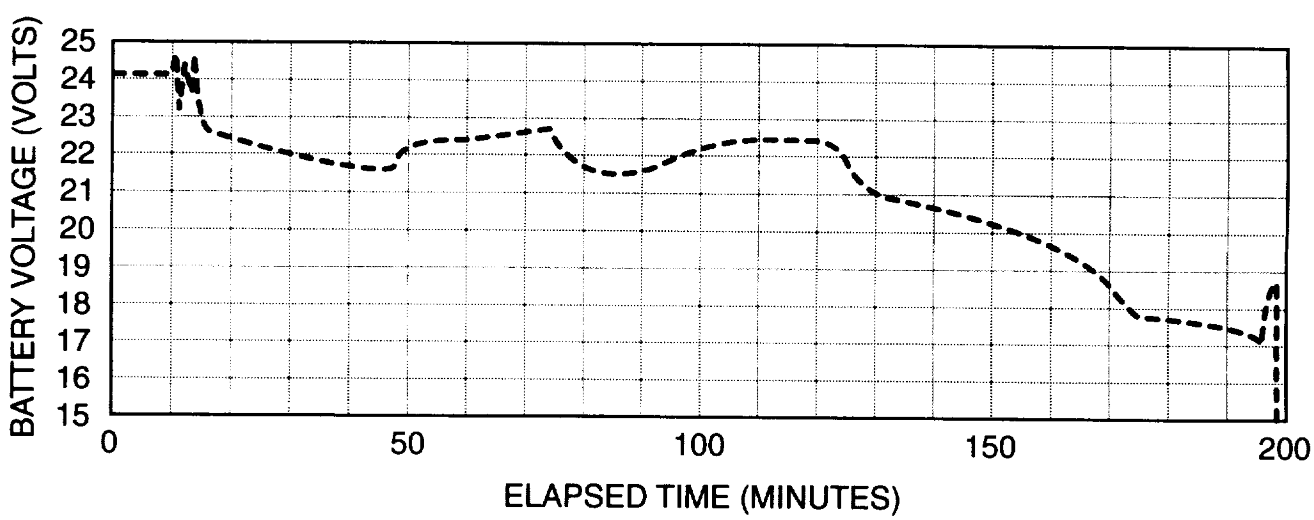


Figure 27. Battery No. 2 discharge (day 274).

10.0 CONCLUSIONS

Battery operations were culminated when the IUE spacecraft's operations were terminated on September 30, 1996. The batteries had successfully provided support to the mission for 38 eclipse seasons over 18 1/2 years, and have gone through approximately 1,080 discharge/recharge cycles. It is noteworthy that this support surpassed the spacecraft design life of 3 years by over 6 times. This longevity can be contributed to several factors:

- excellent process control at the cell manufacturer to activate the cells with a controlled amount of electrolyte, (KOH), 4cc/rated Ah,
- build of cells using teflonated negative plates and dual ceramic seal terminals, and
- meticulous care during battery maintenance and spacecraft operations.

Several attributes should be emphasized such as the IUE Operational Control Center's ability to manually maintain the batteries below 25 °C, control the battery's voltage between 19 and 25 V, and reduce non-critical spacecraft loads during the umbra eclipse periods to reduce battery DOD to help extend the battery life. Observations during the early eclipse seasons indicated that deep battery discharge during the eclipse periods had some reconditioning effects on the batteries. On scheduled intervals the batteries were subjected to shallow discharge and recharge regimes to provide this reconditioning effect, especially just prior to the start of an eclipse season.

Figure 19 shows that load sharing was within 8 percent during the entire mission and that the batteries took turns as the predominant provider, indicative of electro-chemical devices effected by temperature and usage. Considering these factors, the load sharing by the batteries during the entire mission was excellent. Individual battery DOD was close with a maximum delta of six percent. During the final discharge, the end of spacecraft life operations, the capacity of Battery No. 1 was 4.62 Ah and Battery No. 2 was 5.35 Ah. These figures were obtained during spacecraft spinning which created some charge modes as well as the maximum discharge modes created by full residual spacecraft loads. It is, however, remarkable that the batteries indicated above 80 percent of name plate capacity even after 18 plus years of nominal performance.

11.0 LESSONS LEARNED

Even though the IUE batteries provided excellent mission support, some changes from the design to modern technology could have provided even better batteries and would have required fewer IUE Operations Control Center (OCC) manual battery operations.

On IUE spacecraft, Battery No. 2 was mounted adjacent to the external louver with Battery No. 1 mounted to the inside of Battery No. 2 (Figure 1), there was some question of equal heat transfer to the heat pipes. We recommend that future spacecraft designs include the installation of batteries perpendicular to the peripheral of the spacecraft with heat pipes (where applicable) mounted equally below each battery and with batteries equally facing any external louver.

The IUE batteries were mounted to the main platform using corrugated gold foil. Today's technology indicates that mounting batteries to the main platform with a thermal compound such as McGain NuSil or ChoTherm would provide a better thermal interface.

The battery telemetry of some modern day spacecraft includes individual cell voltages to supplement battery voltage. This complementary data provides ample background information to spacecraft analysts who analyze and manage spacecraft batteries' performance. An example would be an observation that one or more cells in a battery show premature loss of voltage and associated capacity. Any anomalies of this type would require immediate corrective action such as reducing spacecraft loads to prevent possible cell reversal. Since the IUE battery telemetry only included battery voltage, current, temperature, and third electrode voltage, the loss of a cell (and possible cell reversal) would have been a calculated guess. Therefore we suggest that incorporating individual cell voltages in battery telemetry would be an excellent design improvement.

In retrospect to the marginal operations of the third electrode charge control systems used on IUE, the use of similar circuits on future spacecraft would require an in-depth analysis of all thermal and electrical aspects of the IUE battery and third electrode charge control circuits to ascertain what design updates and procedural changes were necessary and to correct any abnormalities. The design would have to incorporate technology consistent with the analysis and all components thoroughly life-tested as a system prior to being recommended for space flight.

Three major lessons learned about the third electrode battery charge control circuits are:

- battery laboratory test equipment should have used control circuits similar to the spacecraft electrical circuits (impedance, loading, biasing, etc.),
- battery laboratory test chambers should have been designed to closely emmulated predicted flight thermal conditions, and
- battery mounting and placement was extremely critical for temperature maintenance and uniformity between batteries.

However, we suggest that designing the third electrode control circuits to step the charge current to a trickle low level at a specific third electrode voltage level would be more direct and consistent than the IUE current taper acquired during close loop battery charge operations. The IUE control personnel used the step method to reduce battery charge current to override the current taper process to protect the batteries from overcharging and reduce battery temperatures as required.

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13. ABSTRACT *(Maximum 200 words)*

The "International Ultraviolet Explorer (IUE) Battery History and Performance" report provides the information on the cell/battery design, battery performance during the thirty eight (38) solar eclipse seasons and the end-of-life test data. It is noteworthy that IUE spacecraft was an in-house project and that the batteries were designed, fabricated and tested (Qualification and Acceptance) at the Goddard Space Flight Center.

A detailed information is given on the cell and battery design criteria and the designs, on the Qualification and the Acceptance tests, and on the cell life cycling tests. The environmental, thermal, and vibration tests were performed on the batteries at the battery level as well as with the interface on the spacecraft. The telemetry data were acquired, analyzed, and trended for various parameters over the mission life. Rigorous and diligent battery management programs were developed and implemented from time to time to extend the mission life over eighteen plus years. Prior to the termination of spacecraft operation, special tests were conducted to check the battery switching operation, battery residual capacity, third electrode performance and battery impedance.

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